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PROPULSION REQUIREMENTS FOR SPACE MISSIONS

VOLUME III

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ROCKETDYNE

A DIVISION OF NORTH AMERICAN AVIATION, INC.

6633 CANOGA AVENUE
CANOGA PARK, CALIFORNIA

NAS 5-916

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FOREWORD

This report was prepared in compliance with the provisions of National Aeronautics and Space Administration Contract NAS 5-916, "Research Study to Determine Propulsion Requirements and Systems for Space Missions."

ABSTRACT

Volume III presents the results of Phase 2 of the "Study to Determine Propulsion Requirements and Systems for Space Missions." From the results of the first phase studies, three space missions are selected for consideration, and a booster vehicle is selected for each mission. These three mission vehicle combination are comprehensively investigated.

(Unclassified Abstract)

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CONTENTS

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INTRODUCTION

In this section the results of Phase 2 of the research study to determine propulsion requirement systems for space missions are presented. From the results of the first phase studies three space missions were selected for consideration. A booster vehicle was selected for each mission to establish a nominal spacecraft size. The Phase 2 studies considered these three mission/vehicle combinations in the light of a more comprehensive scrutiny.

The three nominal mission/vehicle combinations selected by NASA for consideration in Phase 2 are described in Table 3-1.

TABLE 3-1

SELECTED MISSION VEHICLE COMBINATIONS

Space Mission	Booster Vehicle	Nominal Earth Escape Payload, lb
Soft Lunar Landing and Return to Earth's Surface (Aero)	Nova H-6	150,000 (Vary from Nova H-8 to Saturn C-2)
Mars Orbit (No Return)	Nova H-6	150,000 (Vary from Nova H-8 to Saturn C-2)
Orbital Rendezvous	Nova H-2	-----

Substituted for C-2

[redacted]

The lunar mission requires a soft lunar landing and return to Earth. Mars orbit mission requirements are the establishment of a circular orbit around Mars with no return. The orbital rendezvous mission requires the mating of two or more vehicles in an Earth orbit in a manner that could be used for orbital buildup of space vehicles. In the selection of these combinations it was understood that primary emphasis would be placed upon the lunar and Mars missions.

The nominal space vehicle for the lunar and Mars missions was based upon the capabilities of the Nova H-6 booster vehicle. Originally the orbital rendezvous studies were to be based upon the capabilities of the Saturn C-2 system. It was understood, however, that the C-2 booster/second-stage combination will not be developed; thus the nominal vehicle was based on the capability of the Nova H-2 booster which is approximately the same as the Saturn C-3. In each of these mission studies the effects of employing alternate booster vehicles which would result in a different size space vehicle were investigated.

Phase 2 study effort is outlined in Fig. 3-1. Preliminary studies of the propulsion systems used in accomplishing the space missions were conducted. These studies included consideration of environmental effects, preliminary selection of engine operating parameters, and estimates of the various deviations in engine performance. Rather than establishing the details of a specific system on detailed engine design layouts, the studies were conducted in a general fashion, being directed toward the preliminary evaluation of the various alternative systems.

The relation of the vehicle propulsion system and the mission/trajectory requirements was directed, of necessity, to receive primary emphasis in the Phase 2 study effort. Although the basic energy requirements of the

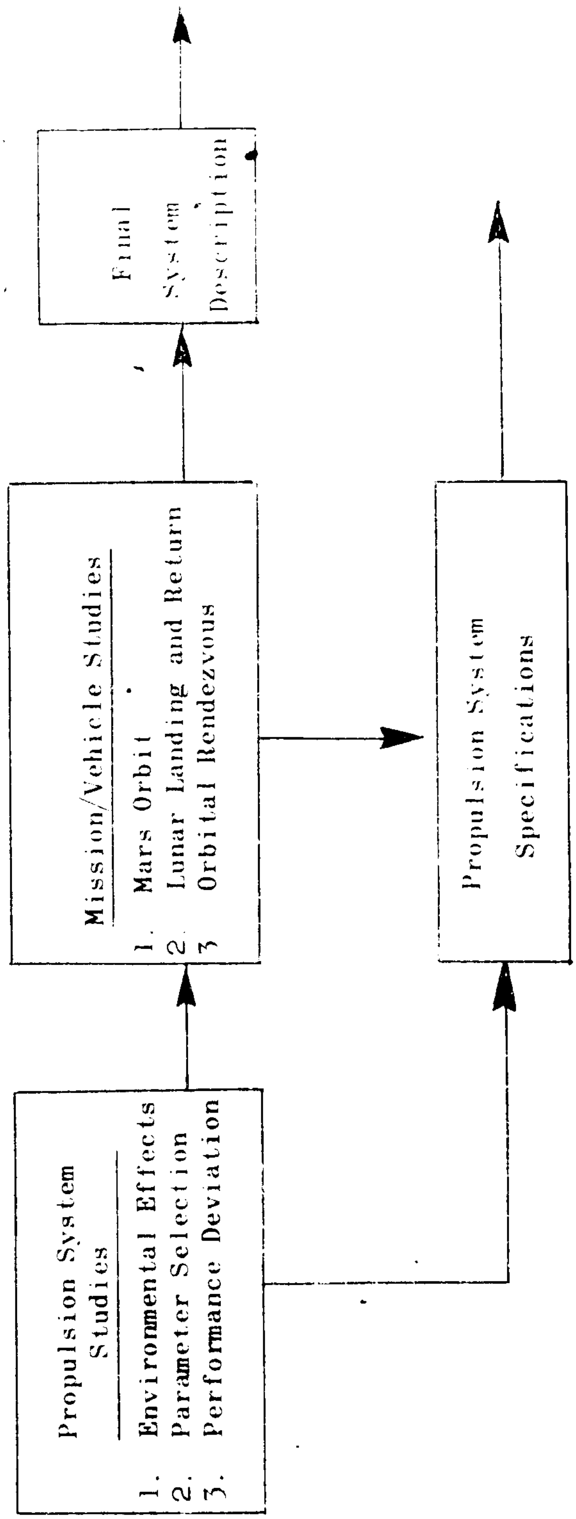

Figure 3-1. Phase 2, Study Effort

various propulsive maneuvers contained in the selected space missions were analyzed in the Phase 1 studies that analysis indicated: (1) the close relationship of trajectory and propulsion requirements, (2) the innumerable alternative trajectory schemes and (3) the lack of any available all-inclusive space mission trajectory information source.

In Phase 2 the individual trajectories for the three space missions were subject to additional detailed evaluation to establish various methods of accomplishing the space missions. A broad spectrum of these maneuver combinations was considered, and the effects of the resulting propulsion requirements on the space vehicle and its payload capability studied. Emphasis was placed on the major propulsive phases. Maneuvers such as attitude control were given secondary consideration. The different system variations, e.g., specific impulse deviation, guidance errors, cutoff impulse deviation were studied in terms of their effects on the propulsion system design and the propulsive trajectory maneuver combinations.

As the technology for space flight advances, specific basic schemes as a result of studies such as this, will be shown to be optimum; however, at the current status continuous further refinement can be expected. Review of ballistic missile technology supports this fact. This study cannot be considered as an end product; thus it is directed to further refine future space propulsion requirements.

As the study is directed to define future propulsion requirements, current restricting criteria which would influence propulsion have not been considered as ultimate criteria. For example, guidance and control systems for interplanetary and soft lunar landing systems must be considered as in their initial phases of development. Improvement in design and concept can be expected. Thus propulsion control requirements dictated by "first flight" systems can not be considered as an ultimate goal. In retrospect, initial ballistic guidance and control systems required the vehicle to "fly" a predetermined line in space, have throttling, and cut-off verniers; improvement in guidance and engine cutoff reproducibility has all but eliminated these requirements.

As a sidelight to these studies yet cognizant of them, the propulsion system characteristics are defined. A list of the specifications necessary to characterize the propulsion systems for the three space missions was developed. Using information from the mission and propulsion system studies, these specifications are described in as great a detail as possible.

The evaluations lead toward the description of recommended systems for the three space missions examined. Realizing that the study effort has not been sufficiently comprehensive to definitely establish the propulsion system details, the recommended systems are considered of a preliminary nature. Areas of study are suggested for future effort which will aid in providing a more detailed definition of the desirable propulsion systems.

The space vehicle systems and their propulsion system are examined to determine possible changes in design concept if alternate booster vehicles are employed which would result in a different size space vehicle system. Booster vehicles in the range Saturn C-2 to Nova H-8 are considered. Effects on staging, feed system, and propellant selection are noted.

SUMMARY AND CONCLUSIONS

PROPULSION SYSTEM STUDIES

The Phase 2 study was conducted to analyze and describe the requirements of the three space missions that were recommended for further study in Phase 1. Necessary to the analysis of all three missions is the study of certain propulsion system features. These features and their effects on the propulsion system can be studied to a great extent independent of a particular space mission.

Space Environment

The various constituents of the space environment were found to have significant effects on the space propulsion system. The conditions of hard vacuum, particulate radiation, zero gravity, meteoroids, and heat transfer are all such that the operation of a propulsion system in space will be seriously compromised unless the proper design procedures are followed.

Hard vacuum and particulate radiation present a problem primarily of material selection. By designing the propulsion system with materials which do not sublimate in a vacuum or deteriorate under particulate radiation, these problems can be circumvented.

Zero gravity and meteoroids present design problems that are receiving considerable attention. Difficulty of separation of gas and liquid in propellant tanks is one of the more significant problems brought about by zero gravity conditions. Numerous methods of accomplishing this have been suggested. Many of these are feasible and can be incorporated in propulsion system design. Protection of propulsion systems from puncture by meteoroids appears to be provided most efficiently by "Whipple meteoroid bumpers." These thin shields surrounding the component to be protected seem to be very effective in reducing the penetration of the high-energy meteoroids. Additional effort is necessary to provide good design information.

Heat transfer in space presents problems in storing propellants for extended times. These problems arise not only from the thermal radiation emitted by the Sun and planets, but from conductive heat transfer which occurs between dissimilar propellants or between the propellants and other internal heat sources. Studies indicate that the "Earth storable" propellants (hydrazine, etc.) can be easily maintained through proper surface and attitude control. For the missions currently contemplated, the cryogenic propellants (hydrogen, etc.) can be maintained by surface and attitude control in combination with the application of a good insulation material such as Linde SI-4.

Missions of long duration may require so much insulation for the cryogenic (high-energy) propellant combinations that a storable propellant system will provide more payload capability. This is illustrated by Fig. 3-2 developed in this study. The figure presents the combination of storage time and propellant weight which cause the payload of the cryogenic (LO_2/LH_2) combination to decrease to that of the storable propellant

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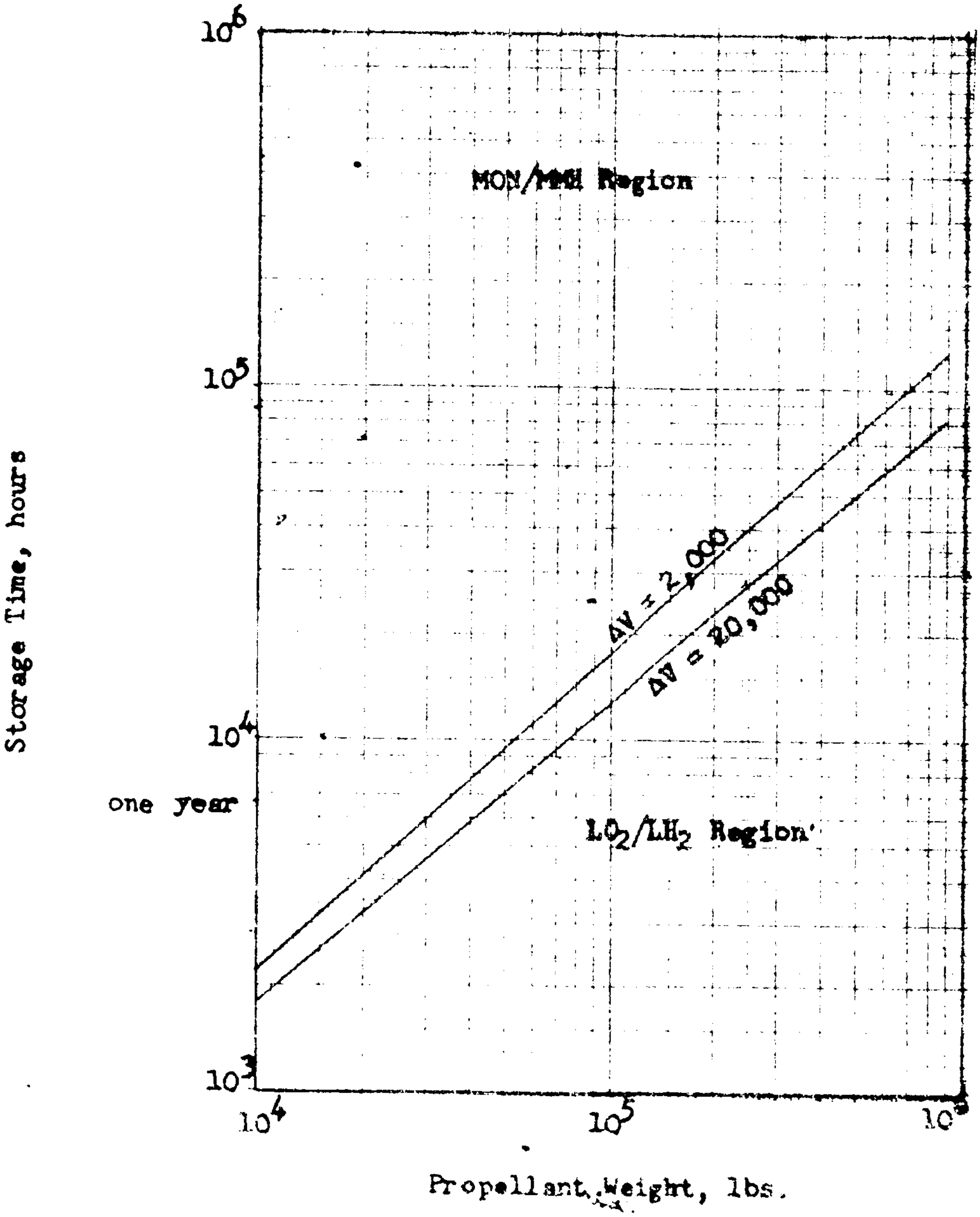

Heat conduction between propellant tanks = 10 Btu/hr
No storage losses
Pump-fed systems
Linde SI-4 Insulation

Figure 7-2. Effect of Space Storage Time on Propellant Selection

[redacted]

(MON/MMH) combination. If the combination of storage time and propellant weight result in a point above the curve, the MON/MMH combination will provide the greater payload; if below the curve the LO_2/LH_2 combination will provide the greater payload.

Studies similar to that above indicated that for the present space vehicle, based on the Nova H-6 booster, LO_2/LH_2 could be used in all stages for both lunar and Mars missions. The storability of these propellants is strongly dependent upon the internal conduction, the size of the vehicle in question, and the method of storage (no loss vs propellant boil-off). These require additional study before a complete evaluation can be made. The internal conduction in particular is a function of the detailed design of the vehicle and is difficult to analyze in a general manner.

ENGINE PARAMETER OPTIMIZATION

It is generally desirable to utilize the propulsion system that provides the maximum payload capability for a given gross weight. This payload capability is strongly a function of the engine operating parameters: mixture ratio, thrust-to-weight ratio, chamber pressure, and expansion ratio. By proper selection of these parameters, maximum payload capability can be provided. Through consideration of previous Rocketdyne studies, parameters were selected for the preliminary propulsion systems to be used in the various mission studies.

Methods are also developed for the rapid evaluation of these operating parameters, given certain hardware information. Using these methods, optimum chamber pressure, expansion ratio, and thrust-to-weight ratio can be determined. The effect of various factors on the optimum chamber pressure is illustrated in Fig. 3-3 for pump and pressure-fed systems. Variation of ±50 percent in certain factors affect the optimum chamber pressure as shown.

ENGINE PERFORMANCE DURING TRANSIENT OPERATION

During engine startup and cutoff, thrust is a function of time. This transient thrust buildup or decay contributes a certain amount of impulse to the vehicle. Due to variations in engine components, this impulse contribution will vary from run to run in a given engine. This variation in impulse can significantly affect the trajectory to be traveled by the space vehicle and must be reduced to a negligible effect through engine system design and/or corrected in a subsequent propulsion phase.

For most propulsive maneuvers the variations in engine start can be taken into account by the guidance system during normal engine operation, and the necessary correction made. Impulse deviations at cutoff were estimated for several propulsion systems as a function of engine thrust. The deviation can be decreased either by lowering thrust or by reducing the main valve closing time. There are, naturally, limits on both of these methods.

Some estimates were made of the effect of this cutoff impulse on the velocity of the space vehicle. These effects were considered for the lunar landing vehicles, and the velocity variation was less than ±1 fps.

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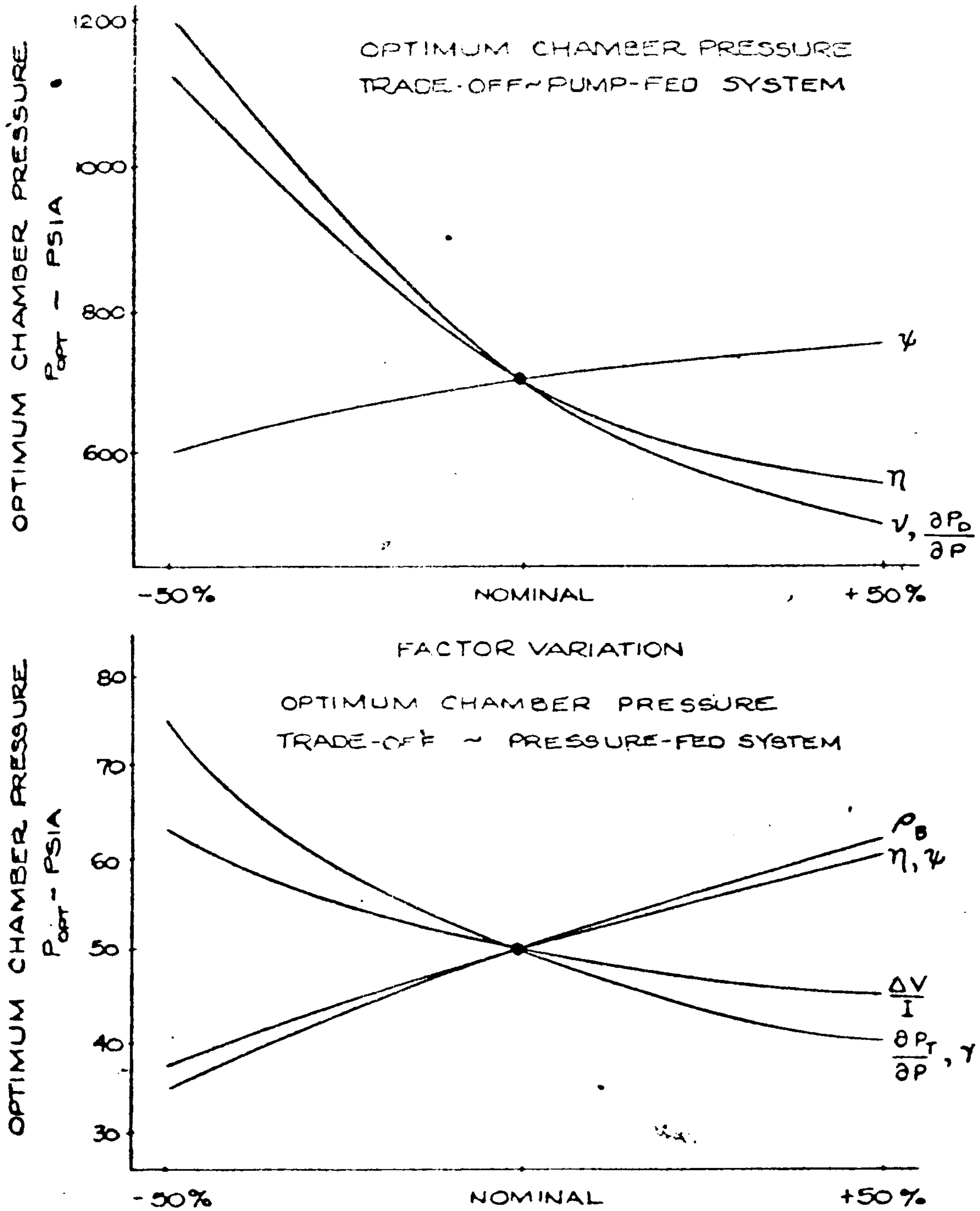

Figure 3-3. Factor Variation

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ENGINE THROTTLING

In several propulsive maneuvers the requirement for engine throttling may arise. This throttling may be accomplished in a single step or may be continuous. The performance losses accompanying throttled operation were studied. In general it was found that the losses in thrust chamber performance (specific impulse) were small for space applications. For example, a case of 5:1 throttling resulted in less than 2-percent decrease in thrust chamber specific impulse at the throttled condition. In a pump-fed system the decrease in engine specific impulse may be greater due to inefficiencies in turbine operation.

SECONDARY PROPULSION SYSTEM CONSIDERATIONS

In addition to the topics discussed above there are some secondary aspects which were considered. Trapped propellant, propellant utilization system, thrust vector control requirements, and vehicle acceleration loads all affect the propulsion system design.

A study was made of the trapped propellant in a LO_2/LH_2, pump-fed vehicle. Trapped propellant is that propellant that is left unburned due to a premature exhaustion of the other propellant. Taking into account off-mixture-ratio tanking and deviation from expected time-average engine mixture ratio operation, the trapped propellant was estimated to be [illegible].78 percent of the usable propellant based on a fuel bias propellant mixing.

For propulsion systems which supply fairly large velocity increments, this amount of trapped propellant will significantly decrease the payload capability. In these systems a mixture ratio control or propellant utilization system would be beneficial.

Analysis of thrust vector corrective torque requirements for space vehicles indicates that generally a gimbal angle of 1 to 2 deg (together with an auxiliary roll control system if a single engine is used) should be employed. Although for some space powered-flight maneuvers a small separate attitude control system would be adequate (thereby allowing a nongimballed engine) consideration of engine thrust vector and vehicle center of gravity misalignments dictates main engine gimbaling. For a specific vehicle these requirements should be analyzed in more detail so that vehicle dynamics can be considered.

Study of the acceleration loads to which a space vehicle would be subjected indicates that the inherent low initial thrust-to-weight systems required would result in low-flight loads during space stage operations: approximately 4 g axial and 0.5 g lateral. More severe requirements are dictated by boost phase and ground handling considerations. Nominal values for these effects are:

Direction	Load, g	Operation Phase
Axial	8	Boost
Lateral	4	Handling

LUNAR LANDING AND RETURN

In the study of the lunar landing and return mission, a large variety of propulsive maneuver combinations were considered. The various schemes considered are described in Fig. 3-4 through 3-7. From this maneuver spectrum two basic methods of accomplishing the lunar mission evolved: (1) direct lunar landing, and (2) orbital landing using an intermediate lunar orbit. These two methods are illustrated in the figures by the shaded regions.

As in all of the space missions, the space vehicle was assumed to be in a 300-n mi Earth orbit. The space vehicle departs from the orbit with a propulsion phase that has the thrust vector aligned with the velocity vector. This phase terminates when the vehicle has attained the energy required to make the Earth/moon transfer in the desired time interval. Midcourse trajectory corrections are considered to be applied in two or more increments to reduce the landing error.

The vertical lunar landing method uses a maneuver which places the vehicle on the lunar surface directly from the transfer trajectory. This is accomplished by means of a fixed-thrust level interrupted burning maneuver in which the velocity and thrust vectors are essentially vertical during the firing. No substantial hovering or lateral translation provisions were included.

The lunar landing from orbit was also studied and is considered to be the most generally desirable type of landing. A 50-n mi orbit is first established (the plane of which is determined by the velocity vector at the beginning of the transfer phase) using a thrust antiparallel to the velocity maneuver. This circular orbit is converted to a 50-n mi/30,000-ft elliptical orbit with the periapsis slightly before the desired landing spot.

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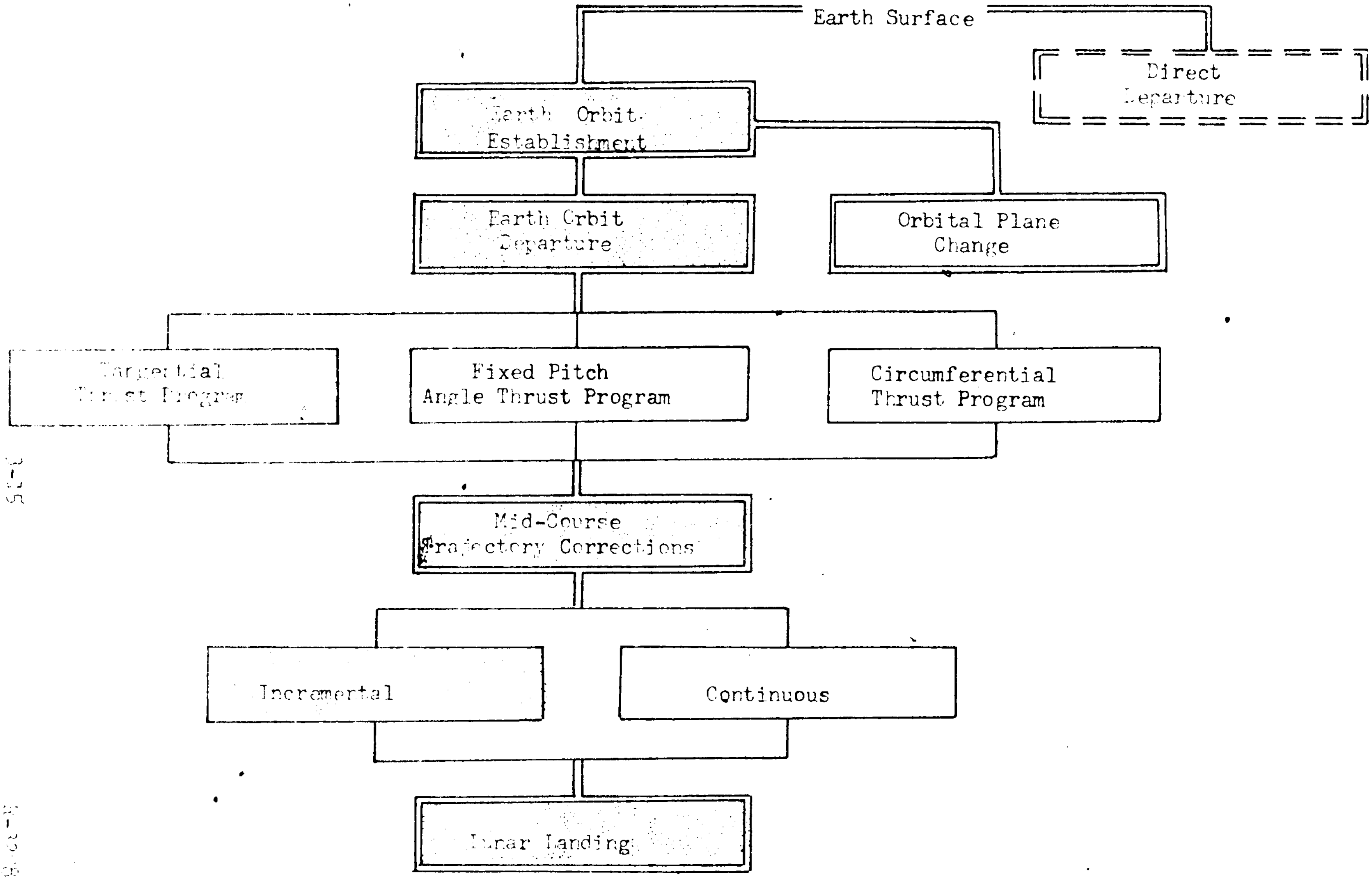

Figure 3-4. Maneuver Combinations for Earth-Moon Transfer

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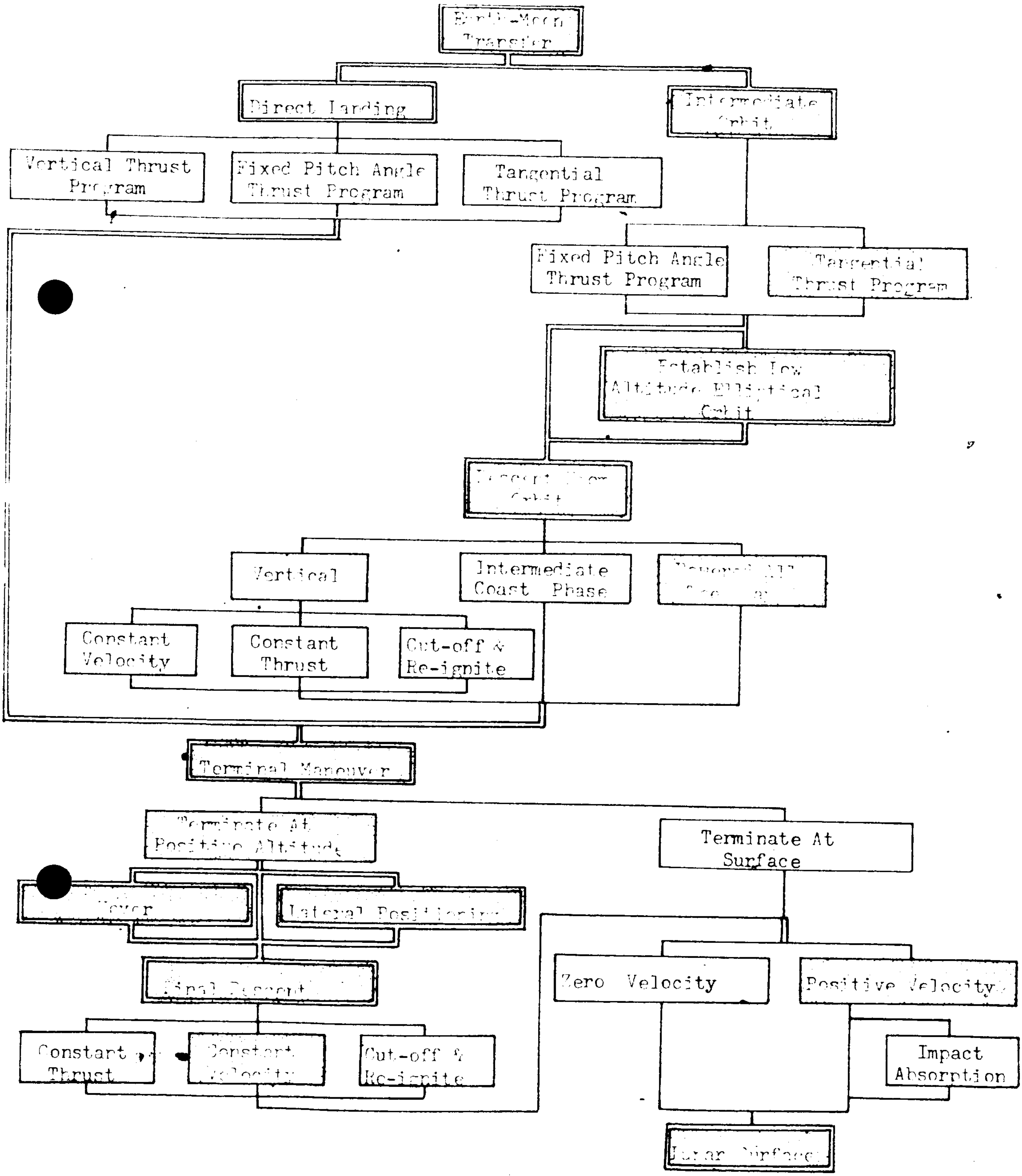

Figure 3-5. Maneuver Combinations for Lunar Landing

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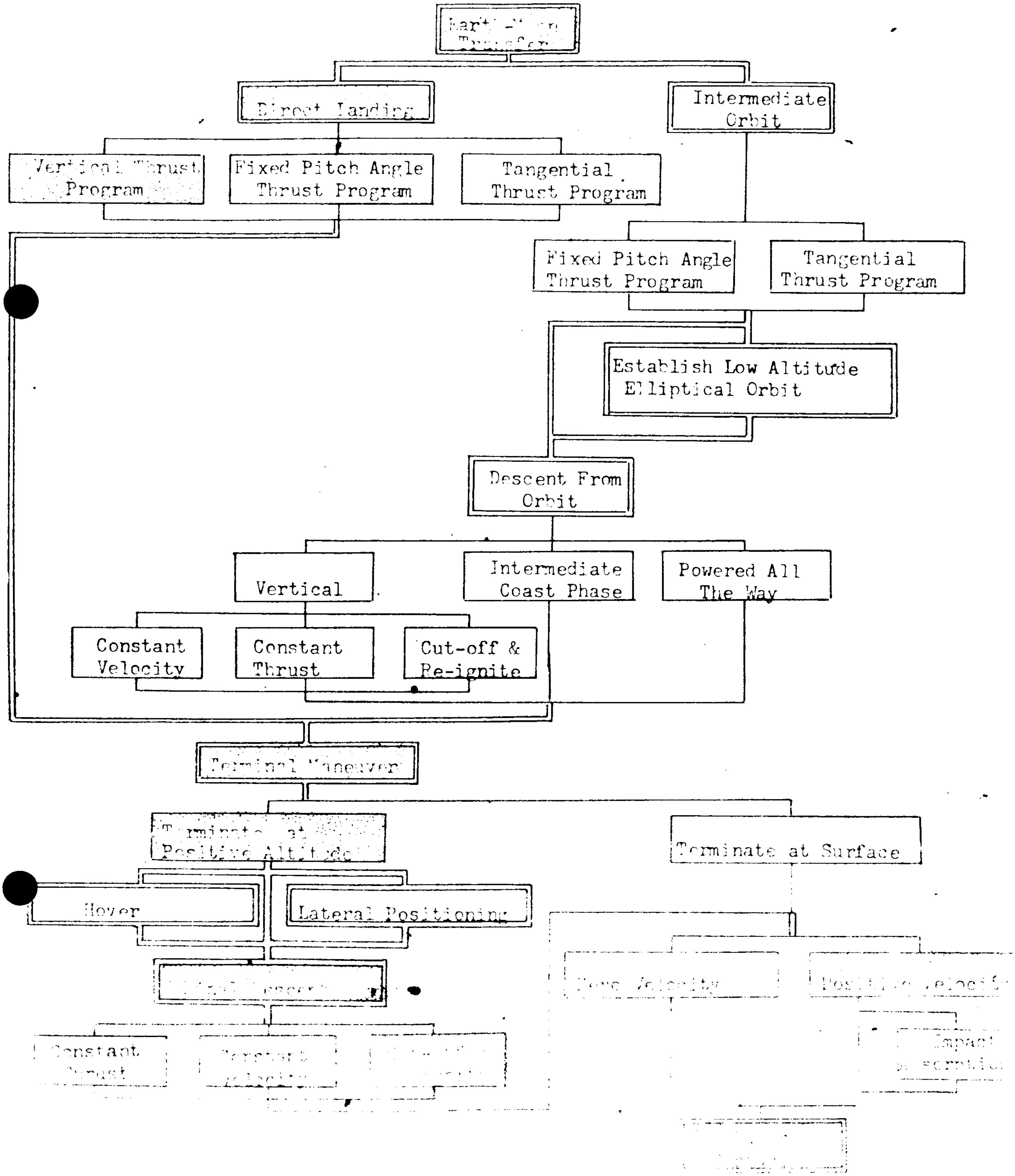

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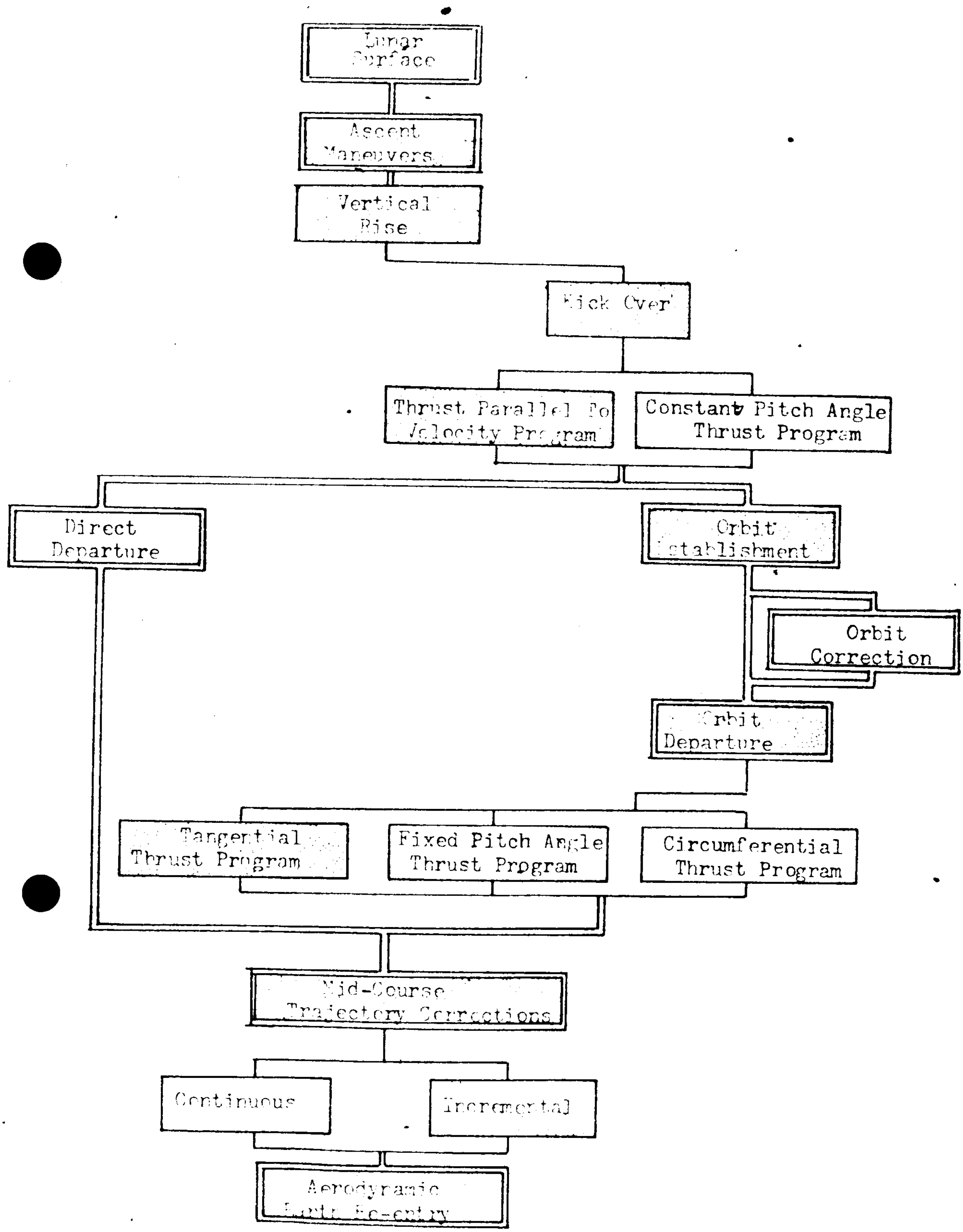

Figure 3-7. Maneuver Combinations for Moon-Earth Transfer

[redacted]

At the periapsis thrust is again applied antiparallel to velocity to bring the vehicle to a low altitude with a small residual descent velocity. Hovering and translational capabilities are provided for this mission.

The takeoff maneuver was determined by the type of landing. Vertical takeoff to moon/Earth transfer trajectory was used in conjunction with the vertical landing maneuver and a takeoff to a 50 n mi prior to the transfer was considered for the orbital landing case.

Midcourse corrections were provided for as in the Earth/moon transfer trajectory. Earth re-entry and landing maneuvers were assumed to be accomplished aerodynamically.

The vertical descent trajectory is most suited to systems having simple guidance systems and fixed-thrust engines. The probability of safe return appears to be lower than that of the orbital trajectory, and the landing point is restricted. Capability of one restart will be required. The orbital landing technique assures that the vehicle will not crash if the engines fail to ignite. The maneuver makes use of (assumed) previous lunar orbital experience and permits landing at any point on the lunar surface.

For the two maneuver combination methods propulsion systems were studied to evaluate engine thrust levels, vehicle staging, and relative payload capability. These studies were based upon a space vehicle weighing 354,000 lb initially placed in a 300 n mi orbit by a Nova H-6 booster vehicle. The effects of using four different propulsion systems were studied. These systems, liquid oxygen/liquid hydrogen (LO_2/LH_2) and mixed oxides of nitrogen/monomethylhydrazine (MON/MMH), represent a broad range of propulsion system characteristics and should indicate the effects of propellant properties on the space vehicle.

The propulsion systems were used in a variety of maneuver/vehicle combinations and were evaluated in terms of performance, complexity, etc. For example, the Earth/moon transfer maneuver is considered using the previously mentioned systems plus a typical solid propellant system. A comparison of these vehicles shown in Fig. 3-8, clearly indicates the advantages of the LO_2/LH_2, pump-fed system.

From consideration of a large number of combinations, the vehicles recommended for use in the two lunar missions were selected. These are described in Tables 3-2 and 3-3. Pump-fed systems were selected over the pressure-fed systems as they provide a significant payload advantage due to lighter engines and tanks. Propellant storage studies indicated that the LO_2/LH_2 combination could be easily maintained for the lunar missions contemplated. Thrust levels selected are near optimum for the maneuver-stage combination selected. However, a wide variation in thrust is possible without severe payload loss, e.g., the J-2 engine (200,000 lb thrust) may be used as the first stage propulsion system in either vehicle without affecting payload appreciably.

The two-stage vehicles were selected since it was felt that their simplicity was of more benefit than the slight payload increases achieved with a greater number of stages. Other considerations, such as to the provision for abort capability at all times during the mission, could modify the staging selection.

Staging on the lunar surface is preferable for the vertical descent trajectory because it permits use of a previously fired propulsion system for the critical landing phase, and has the added advantage of protecting the takeoff propulsion system from impact damage upon touchdown. For the

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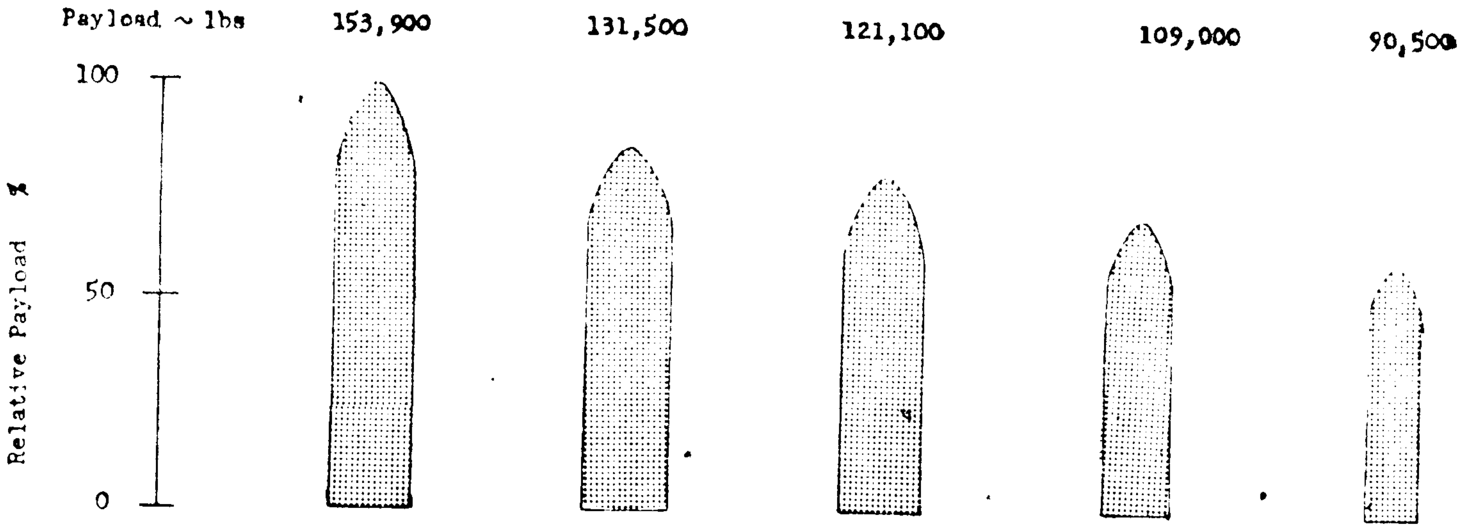
Initial Gross Weight in 300 n. mile Earth Orbit = 354,000 lbs.
(Based on NOVA H-6 Capability)



Propellants	LO_2/LH_2	LO_2/LH_2	MON/MMH	MON/MMH	SOLID
Feed System	Pump	Pressure	Pump	Pressure	--
Thrust lbs	125,000	88,500	127,000	97,000	250,000
p (Calc.)	0.927	0.876	0.950	0.912	0.900

Figure 3-8. Earth Moon Transfer Propulsion 2.6 Day Coast Time

[redacted]

[redacted]

TABLE 3-2

LUNAR LANDING AND RETURN VEHICLE USING AN
INTERMEDIATE LUNAR ORBIT

Payload Weight on Moon/Earth Transfer, lb	29,500
Stage Two	
Propellants	LO_2/LH_2
Feed System	Pump
Throttling	6:1 Step 6 Percent Continuous
Restarts	3
Gross Weight, lb	116,000
Thrust, lb	91,000
Number of Engines	7
(1 Redundant; 1 Throttleable)	
Stage One	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	1
Gross Weight, lb	354,000
Thrust, lb	125,000
Number of Engines	1

[redacted]

TABLE 3.3

RECOMMENDED SYSTEM FOR DIRECT LUNAR
LANDING AND RETURN MISSION

Payload Available for Earth Re-entry, lb	26,300
Stage Two	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	None
Gross Weight, lb	37,500
Thrust, lb	56,000
Stage One	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	1
Gross Weight, lb	354,000
Thrust, lb	248,000

orbital landing, it is desirable to stage prior to the descent from orbit maneuver in order that the thrust required for that maneuver does not unfavorably influence the thrust level selection of previous maneuvers. Preliminary review tends to indicate a redundant multiengine propulsion system should be used for increased landing reliability.

A broadband (approximately 5 deg) three axis, attitude control, propulsion system which functions during the entire transfer can be included at a weight of less than 100 lb. The midcourse correction and orbital conversion maneuvers can be performed by the main propulsion system using the attitude control engines for propellant settling

MARS ORBIT ESTABLISHMENT

The recommendations and conclusions of the Mars orbit mission studies of Phase 2 can be divided into two categories: those conclusions relative to the propulsion/vehicle system, and those pertaining to maneuvers. The separation does not imply independence of the parameters within the broad categories.

A variety of propulsive maneuver combinations were considered for this mission. These are indicated in Fig. 3-9 The maneuver combinations for accomplishing this mission were selected after analysis and review. These are indicated by the shaded areas in the figure. Analyses of Earth-Mars interplanetary trajectories, based on simulated elliptical planetary orbits, indicate a minimum energy launch period occurs about once every two years, and results in Earth-Mars transfer times of approximately 170 to 240 days dependent upon the year of launch.

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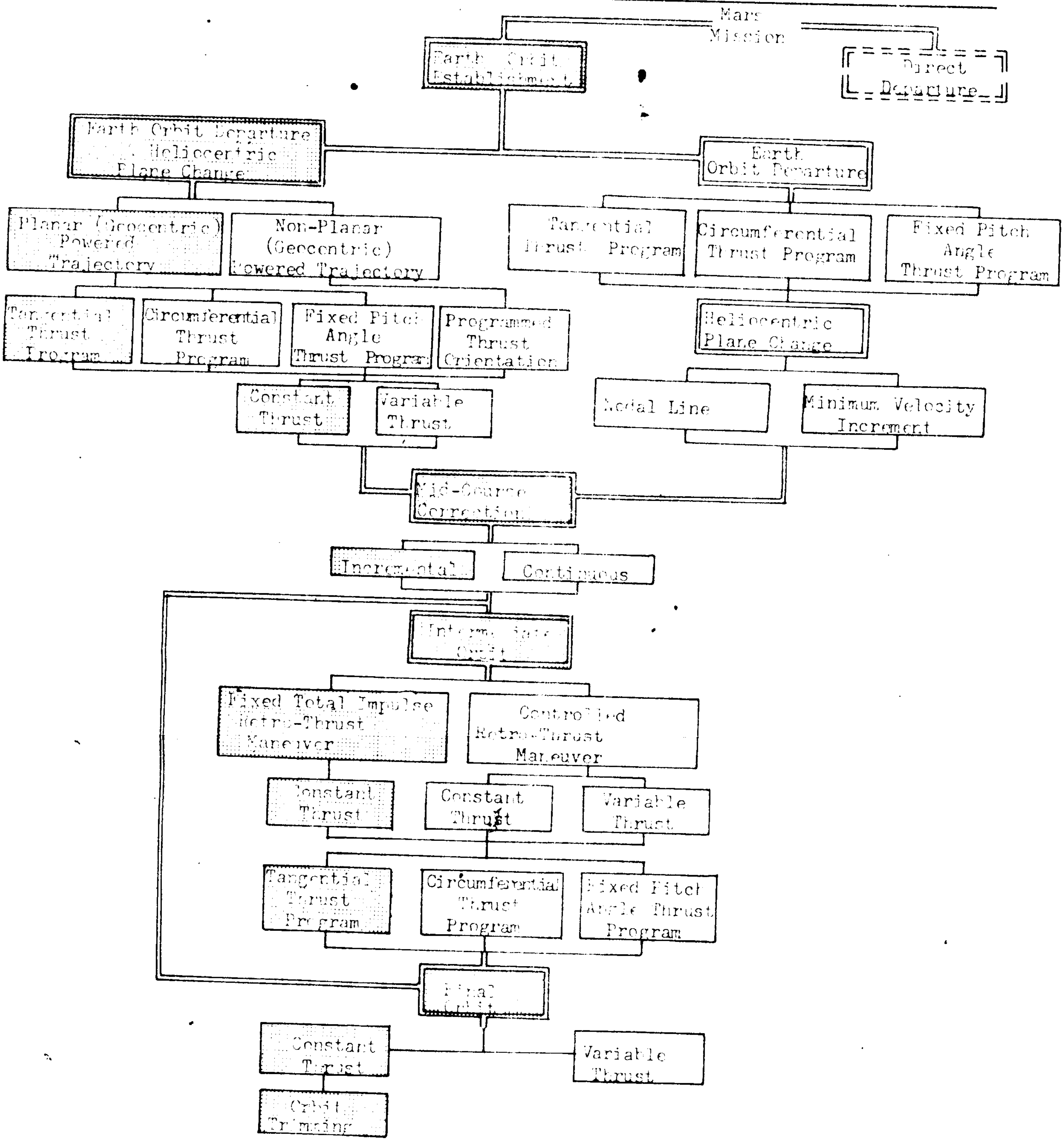

Figure 3-9. Maneuver Combination for Mars Circular Orbit Establishment

[redacted]

During a minimum energy period the space vehicle departs from an earth orbit inclined to the equator to permit a planar propulsion phase. The thrust vector is aligned with the velocity vector during the propulsion phase. The propulsion phase terminates after the vehicle has attained the energy requirements of the particular launch date.

The midcourse corrections are applied in two increments. The first is applied after a 20-day delay from launch and the second shortly prior to planetary intercept. The second or terminal correction for establishing the entry corridor (entry corridor correction) provides the desired asymptotic approach distance at Mars. Both corrections modify the trajectory to maintain a constant transfer time.

A Mars capture maneuver that employs the pre-established asymptotic approach distance is recommended. The retrothrust propulsion maneuver begins at an altitude determined by the hyperbolic approach velocity of the vehicle with respect to the planet. An intermediate orbit is established to ensure capture by the planetary gravitational field. The intermediate orbit is corrected by Hohmann type maneuvers to the final recommended 300 n mi circular orbit.

Propulsion system studies were conducted for this mission using a 334,000-lb space vehicle placed in a 300 n mi Earth orbit by a Nova II-6 booster vehicle. The natural velocity increment separations and storage periods involved indicate that a separate stage for each of the two major propulsion phases (Earth and Mars) be employed. The space vehicle is designed to be capable of launch at anytime during a one-month interval of Earth orbit departure dates which occur at approximately two-year intervals. The stages are filled with the propellant requirements of the launch date. The flexibility permits the basic vehicle to be applied to the optimum launch periods of a number of years. The space vehicle is described in Fig. 3-10.

The first stage for Earth orbit departure uses LO_2/LH_2 as propellants. The propulsion system operates at a constant-thrust level and is a pump-fed system. The thrust level for the nominal vehicle has been selected to be 150,000-lb thrust. Based on the engine grouping in Phase 1 of this study, the propulsion system of this stage could be two 75,000-lb-thrust engines. However, if preferred, the currently developed 200,000-lb-thrust (J-2) O_2/H_2 engine could be used with very little performance change. This performance change is demonstrated in Fig. 3-11, which shows the effect of the first stage thrust magnitude upon the stage payload-to-gross weight ratio. The figure also shows the performance change for a selected thrust magnitude when the initial gross weight of the space vehicle is changed. This illustrates the application of the propulsion system to other vehicles.

For the nominal 354,000-lb vehicle, a second-stage thrust level of 30,000 lb is recommended. Figure 3-12 presents the payload-to-gross weight ratio for the second stage as a function of the thrust magnitude and the initial gross weight. It shows the performance change of a propulsion system operation with a change in the initial gross weight, and the performance change for a selected vehicle with a thrust magnitude change.

For this stage, LO_2/LH_2 propellants are feasible. If the initial gross weight of the space vehicle decreases significantly, storable propellants appear more advantageous. As the initial gross weight of the space

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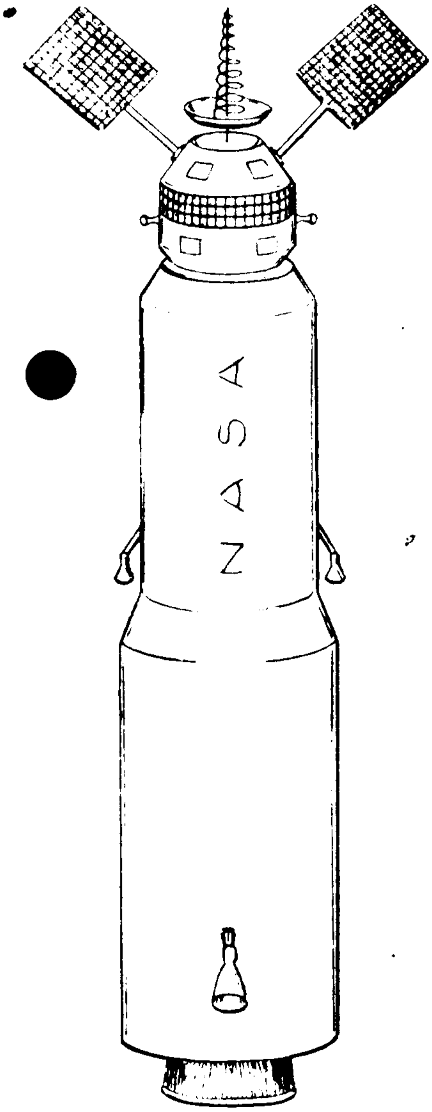

Gross Weight: 354,000 lbs.

Payload: 34,480 lbs.

Stage Two:*

Thrust: 30,000 lbs.
Propellants: LO_2/LH_2
Propellant Tanks:

	Design Capacity	Loading Variations
LO_2 :	13290 lbs.	13,290 - 8,330 lbs.
LH_2 :	66,450 lbs.	66,450 - 41,650 lbs.

Turbopump Fed

Stage One:

Thrust: 150,000 lbs.
Propellants: LO_2/LH_2
Propellant Tanks:

	Design Capacity	Loading Variations
LO_2 :	39,250 lbs.	35,130 - 39,250 lbs.
LH_2 :	196,250 lbs.	175,650 - 196,250 lbs.

Turbopump Fed

* Stage designed to establish intermediate orbit and to restart to change to final orbit.

Figure 3-10. Recommended Two Stage Vehicle for Mars Orbit Mission

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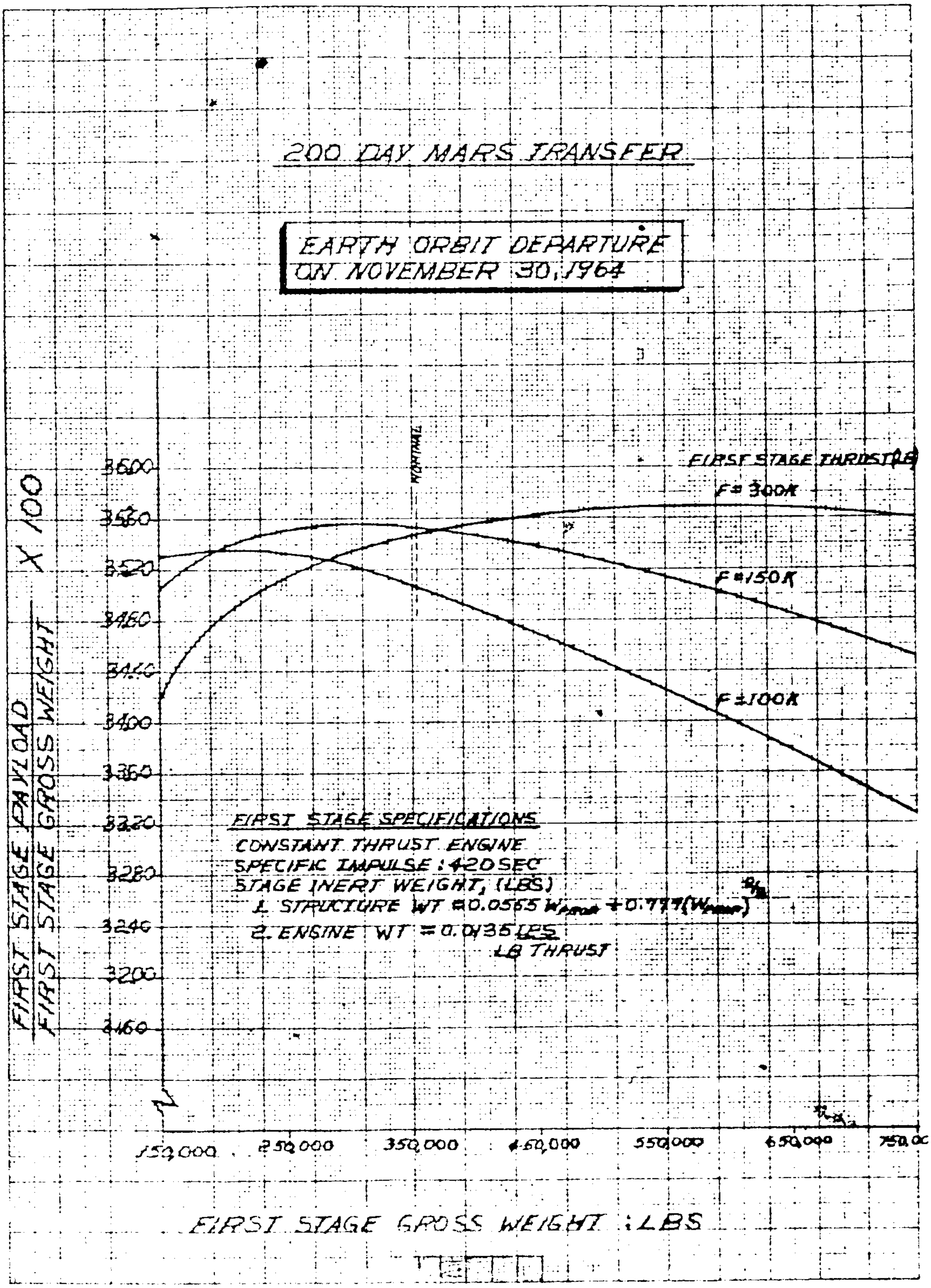

Figure 7-11 200 Day Mars Transfer
First Stage Specifications

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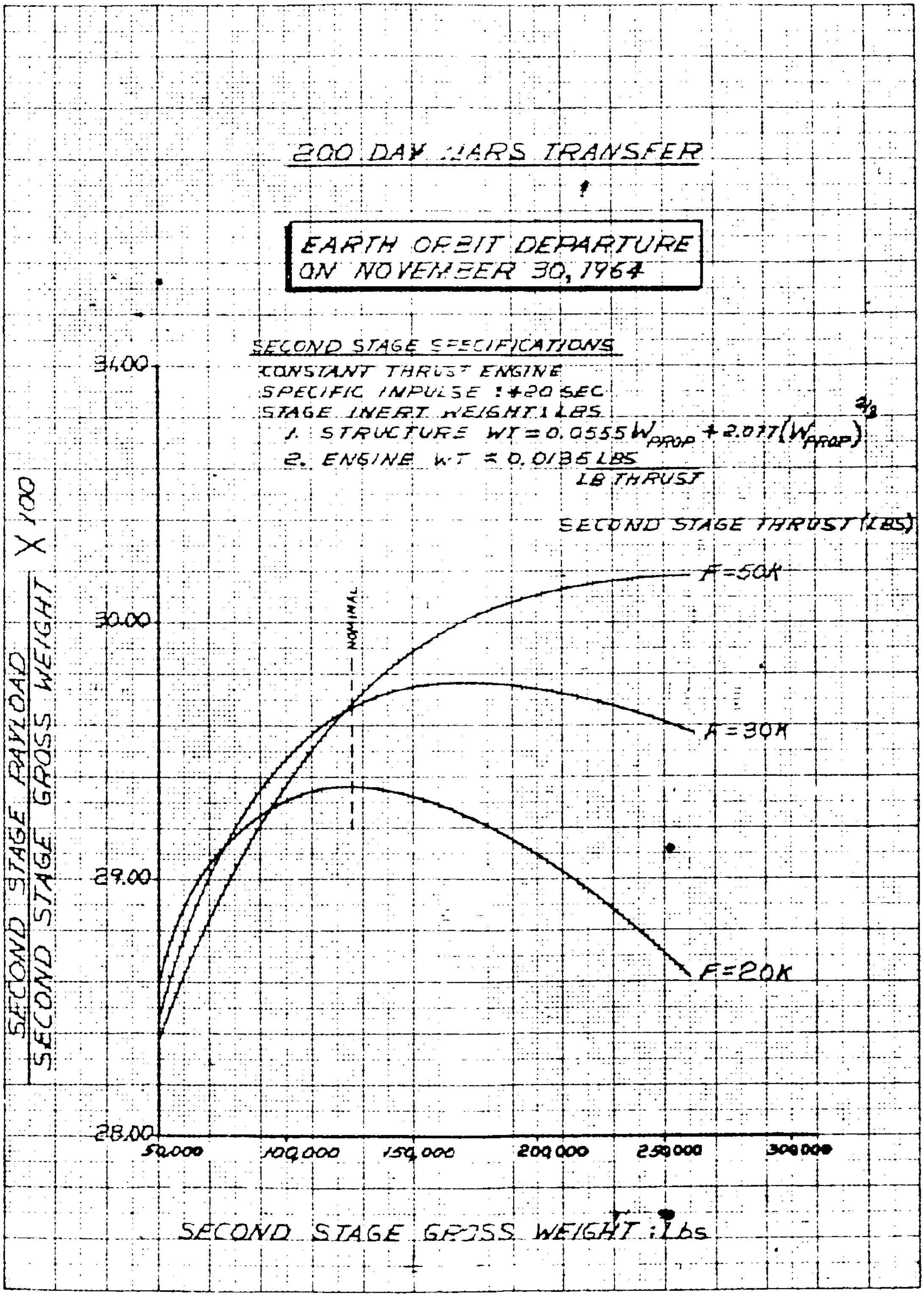

Figure 3-12. 200 Day Mars Transfer
Second Stage Specifications

[redacted]

vehicle increases, the cryogenic propellants appear definitely more favorable. The second-stage propulsion system is recommended as a cluster of three 10 000-lb-thrust engines operating at a constant-thrust level. One of the engines must be restartable to change the intermediate orbit into the final orbit. The propulsion systems will be pump-fed.

The midcourse corrections should be applied by an independent system with a capability of approximately 300 ft/sec total velocity increment. This system will be external to the sealed second-stage engine system. The mid-course correction system may be an integral part of, or the totality of the system required for attitude control of the vehicle during the transfer phase. Since attitude control system analyses were beyond the scope of this study, no recommendations can be made as to the integration or separation of the midcourse propulsion system with the attitude control system.

EARTH ORBIT RENDEZVOUS

In the study effort of Phase 2 the Earth orbit rendezvous was to be considered in a secondary manner compared to the attention given the lunar and Mars missions. This effort was conducted placing considerable emphasis on the material available in the literature and the Phase 1 studies.

A large number of methods of accomplishing rendezvous are available. From consideration of the various combinations and the accuracy of current booster vehicle systems, the following maneuver and staging were selected. A conventional boost maneuver is accomplished by the first and second stages of the booster vehicle. Following the coast to apogee, the final stage fires to establish an orbit. Upon approaching the desired orbit (and target) the final stage is reignited to accomplish the required plane change while leaving a small residual closing velocity between the target and vehicle. Multiple on-off operation of the final stage is used to achieve rendezvous, with the attitude control system used to perform the actual docking maneuver. The final stage may be used in application of retrothrust to initiate aerodynamic re-entry, if required. This sequence appears attractive on the basis of reliability, guidance requirements, and payload considerations.

A restartable, fixed-thrust level, pressure-fed, storable propellant system having a thrust-to-weight ratio in the order of 0.1 is recommended on the basis of reliability, with consideration also given to payload capability and guidance requirements. The payload sensitivity to various propulsion systems is low due to the small velocity increment involved. The selected system is shown in Table 3-4 based on an H-6 booster and a 300 n mi orbital mission including a 5-deg plane change.

It is suggested that strong emphasis be placed on the operational aspects of the rendezvous mission since these may be a major factor in determining the mission characteristics.

TABLE 3-4

RENDEZVOUS PROPULSION SYSTEM

Payload, lb	92 800
Propulsion System	
Feed System	Positive Expulsion
Propellants	MON/MMH
Propellant Weight, lb	28 400
Inert Weight, lb	3200
Thrust, lb	12 000
Restarts	3

PROPULSION SYSTEM STUDIES

INTRODUCTION AND GENERAL DISCUSSION

A number of factors affect the design of the space vehicle through their effects on the vehicle propulsion systems. This section considers space propulsion systems and the effect of some of these factors on them. The effects can in turn be related to the space mission and vehicle design.

One of the most significant of these factors is the space environment. Environmental constituents such as thermal radiation and meteoroids exert a strong influence on propulsion system design, particularly with respect to propellant selection.

It is generally desirable to select the propulsion system that provides the highest payload capability. The propulsion system operating parameters such as mixture ratio, chamber pressure, etc. have a significant effect on the payload capability of a propulsion system. Proper selection of these parameters is an important feature of the space vehicle design.

In space missions it is important that energy is applied to the vehicle in fairly precise quantities. The precision of this energy application is determined by the manner in which the propulsion system is cut off. The precision of the engine cutoff and its relation to the trajectory are important to the space vehicle design.

Certain propulsive maneuvers in space will require throttling. The effects of any inefficiencies resulting from this throttling must be considered in space vehicle design.

All of these factors are considered in detail in this section and their effects on the propulsion system noted. Considering these effects a preliminary description of some propulsion system types that might be used in space was formulated. These preliminary systems will be used in the study of the space missions.

In addition some factors which are secondary in their effect on the propulsion system are discussed. Although secondary in nature these effects must be considered to provide a good vehicle design.

OPTIMUM ROCKET ENGINE PARAMETERS FOR VACUUM OPERATION

In the design of propulsion systems it is generally desirable to select the engine operating parameters which will result in the highest amount of payload weight for a given system gross weight. Although secondary in importance from an over-all vehicle standpoint, four engine operating parameters significantly affect space vehicle performance: (1) mixture ratio, (2) chamber pressure, (3) expansion ratio, and (4) thrust-to-weight ratio. It is the purpose of this study to discuss the selection of these engine operating parameters. The parameter selection briefly described in the Phase-I analysis is expanded. In addition, a method for preliminary selection of some of these parameters was established. This is described along with some of the factors which influence the values selected for these parameters. Final establishment of the engine/vehicle system description and, in particular, the evolution of a detailed component weight description will allow a more definite optimization of these engine operating parameters. The method described and the review of previous Rocketdyne optimization studies provides good preliminary information and allows discussion of the various influencing factors.

Optimum mixture ratio selection is accomplished through a consideration of the tradeoff between specific impulse and propellant bulk density. The optimum usually occurs somewhat higher than the maximum specific impulse mixture ratio because of the effects of bulk density on propellant tanks and feed system weight. The deviation of the optimum mixture ratio from the maximum specific impulse mixture ratio value is particularly evident where one propellant has a low density and, where the tank and feed system weights are high. This effect for the liquid oxygen/liquid hydrogen

propellant combination is illustrated in Fig.3-13. Various specific tank
weights are considered. For the lowest specific weight value, the opti-
mum mixture ratio was 5.45. For the highest value the optimum is
approximately 6.0. Values of optimum mixture ratio for the propellant
combinations considered were taken from previous Rocketdyne studies.

Chamber pressure optimizations consider the tradeoff between thrust chamber
weight, feed system weight and specific impulse. For pressure-fed systems
the optimum chamber pressure is a function of propellant bulk density.
For pump-fed systems the effect of chamber pressure on payload capability
is slight over a fairly large range of chamber pressures. Chamber pres-
sures values were determined from a review of previous Rocketdyne studies.

Optimum expansion area ratios consider the tradeoff between nozzle weight
and specific impulse. For vacuum operation the optimum expansion area
ratios are usually high. This is illustrated in Fig. 3-14. Diameter limi-
tations of a particular vehicle may result in the expansion ratios actu-
ally used in a vehicle being considerably smaller than optimum. Values
of expansion ratio were selected from consideration of previous studies.

Thrust-to-weight ratio optimization depends upon the tradeoff between
engine weight and ideal energy requirements. The ideal energy require-
ments are affected by the thrust-to-weight through variations of the
gravity losses occurring in the propulsive phase. An illustration of
this is shown in Fig. 3-15. The engine weight increases relative to the
remainder of the system as the thrust-to-weight is increased. Ideal
energy requirements decrease as thrust-to-weight increases due to the
decrease in burning time, and thus the gravity losses. The results of

[redacted]

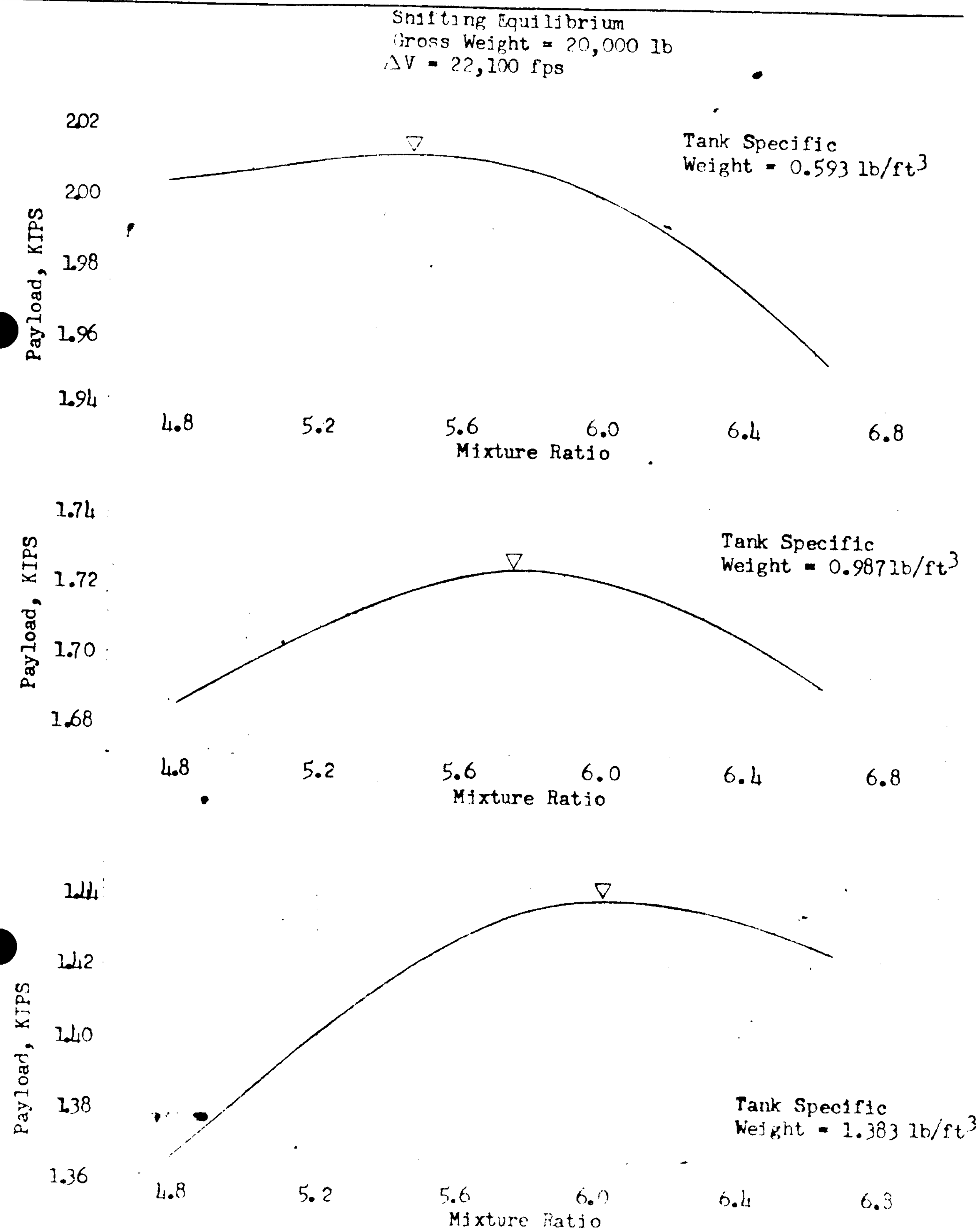

Figure 3-13. Mixture Ratio Optimization for LO_2/LH_2

[redacted]

LO_2/LH_2
P_c = 650 psia
O/F = 4
F = 150 K
100% Frozen Equilibrium Specific Impulse

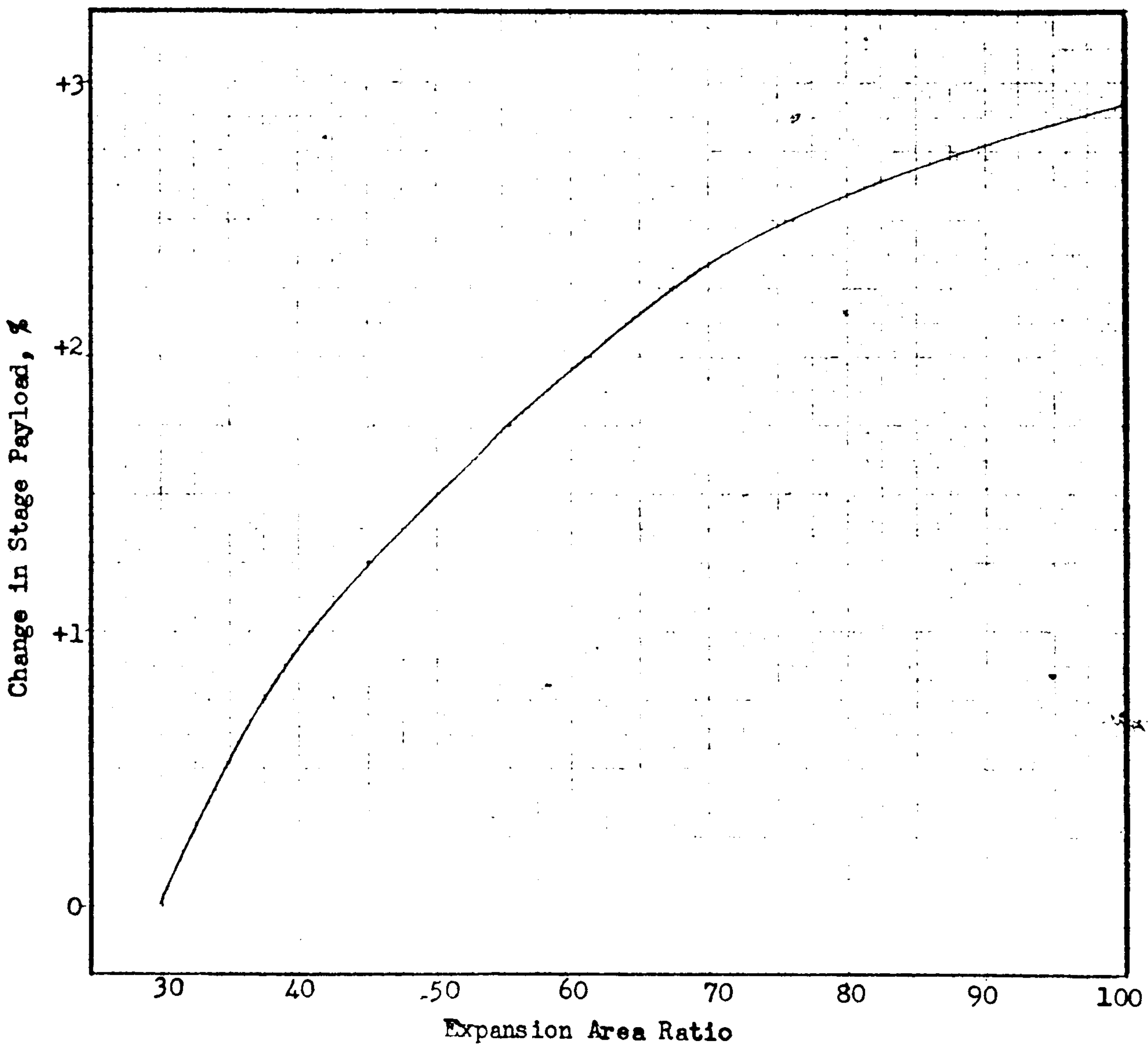

Figure 3-14. Effect of Expansion Area Ratio on Payload Capability

[redacted]

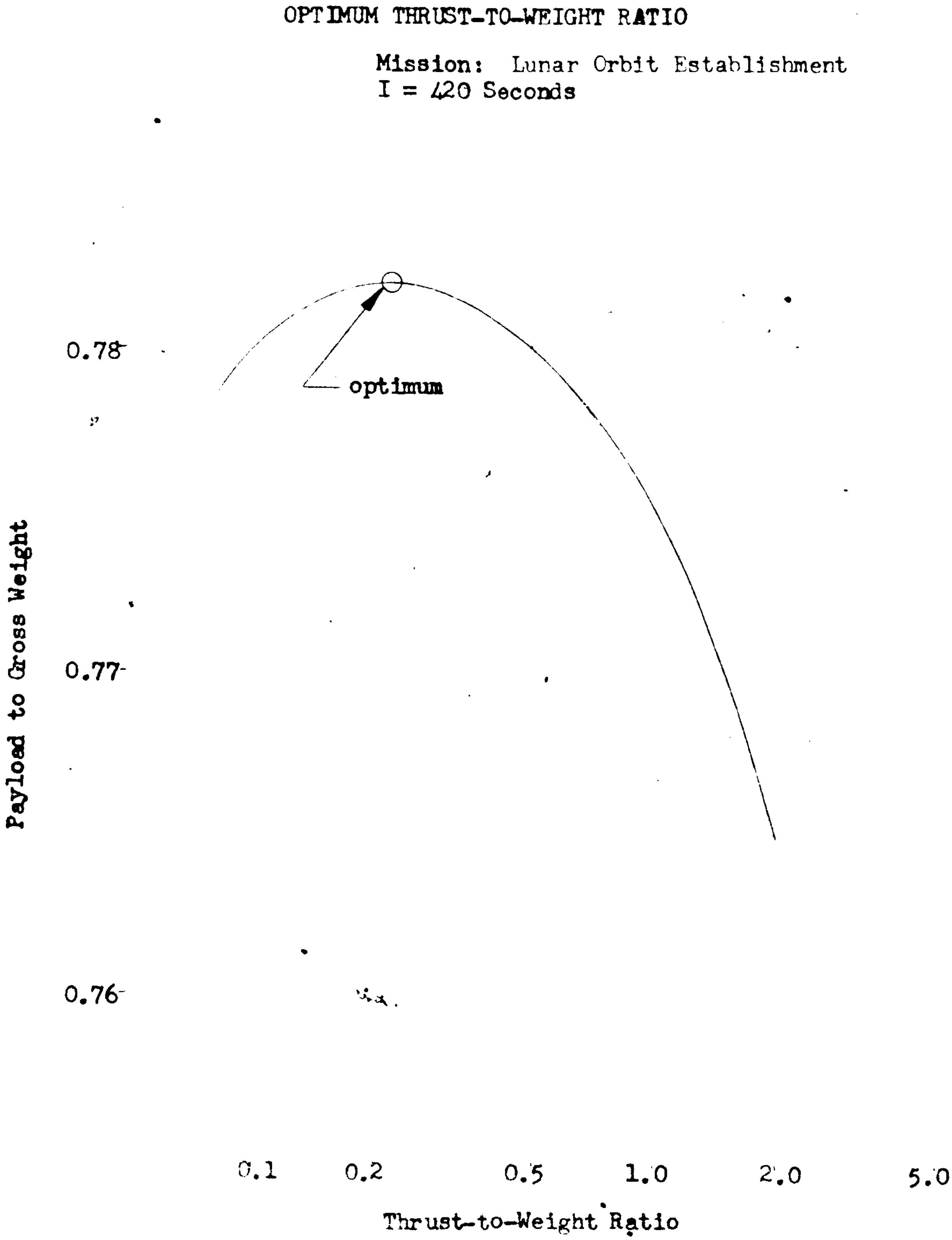

Figure 3-15. Optimum Thrust-to-Weight Ratio

[redacted]

these effects is shown in Fig 3.5 where payload-to-gross weight ratio is plotted as a function of thrust-to-weight ratio. It can be seen that although in the range of thrust-to-weights considered the variation of payload-to-gross weight ratio is small a fairly definite optimum occurs. Optimum thrust-to-weight ratios for the missions considered were selected from previous Rocketdyne studies and a method described in subsequent portions of this report.

The operating parameters for four types of propulsion systems are considered. These systems provide a large range of propulsion system properties and serve to indicate the effects of these properties on the mission and vehicles.

Propellant	LO_2/[illegible]	LO_2/LH_2	MON/MMH	MON/MMH
Feed System	Pump	Pressure	Pump	Pressure
Chamber Pressure, psia	500	60	500	150
Expansion Ratio	30	16	30	25
Mixture Ratio	5.0	5.5	2.4	2.4

These operating parameters are based on previous Rocketdyne studies but should provide near optimum results for the systems considered. A better idea of the optimum parameters can be had from a more definite description of the propulsive requirements and component weights, using a detailed system optimization.

[redacted]

Estimation of Optimum Engine Parameters

To provide methods for quickly estimating optimum engine operating parameters the following analyses were conducted. These methods allow estimation of (1) optimum expansion ratio, (2) thrust-to-weight ratio, and (3) chamber pressure for vacuum operation.

In optimizing engine operating parameters, the usual method of parameter selection is to vary the parameter over a given range, plot the resulting payload weight, and determine the maximum value from this plot. For vacuum operation which is characteristic of space flight a somewhat more explicit method can be used. Assumption of vacuum operation eliminates the necessity of considering the effect of ambient pressure on rocket engine performance.

For a single propulsion stage the payload-to-gross weight ratio can be expressed in terms of energy requirements, engine performance, and various structural factors. The latter two factors are functions of the engine operating parameters. In this study three parameters were considered: (1) expansion ratio (ϵ), (2) initial thrust-to-weight ratio (η), and (3) chamber pressure (P). The parameter values giving maximum payload are obtained by assuming structural factor relations, setting the payload derivative equal to zero, and solving the resulting equation.

Analysis. The mass-ratio of a single-stage vehicle can be expressed as

$$R = \frac{1}{\mu + \left(\frac{R-1}{R}\right) + f\,\eta}$$

[redacted]

where

R = mass ratio

μ = payload weight/gross weight of structure

k = specific weight of structure proportional to propellant weight ~ lb structure/lb propellant

f = specific weight of structure proportional to thrust ~ lb/structure/lb thrust

η = initial thrust-to-earth weight ratio ~ lb thrust/lb gross weight

Solving for the payload-to-gross weight ratio

$$\mu = \frac{1}{R} - k \left(\frac{R-1}{R}\right) - f\,\eta \qquad (1)$$

The mass-ratio can also be expressed

$$R = e^{\Delta V/c^{x} C_F} \qquad (2)$$

where

ΔV = ideal velocity requirement of mission ~ fps

c^{x} = characteristic velocity ~ fps

C_F = thrust coefficient

Optimum Expansion Ratio

For the optimum expansion ratio determination the following assumptions were made

1. c^*, k, η, V = constant
2. f, C_F, = functions of ϵ
3. μ to be maximum
4. conical nozzle

For the conical nozzle a relation between the engine specific weight factor (f) and expansion ratio was developed. Derivatives of this relation as well as Eq. 1 and 2 were taken with respect to ϵ. These derivative equations were solved letting $\partial\mu/\partial\epsilon = 0$ (Appendix A)

$$\left[\frac{\Delta V}{c^x}\right] e^{-\left[\frac{\Delta V}{c^x}\right]\frac{1}{C_F}} \left[\frac{1}{C_F}\left(\frac{\partial C_F}{\partial \epsilon}\right)\right] = \left(\frac{\eta}{1+k}\right)\frac{\rho t}{P \sin \alpha}$$

where

ρt = nozzle weight factor ~ lb/sq in.

α = nozzle half-angle

The relation of C_F and $\frac{\partial C_F}{\partial \epsilon}$ to η are very involved and are best determined numerically. This was done for a combustion product specific heat ratio of 1.23 (typical of liquid oxygen/liquid hydrogen and liquid oxygen/RP propellants).

Assuming a nozzle half angle (α) of 15 deg and the specific heat ratio of 1.23, the optimum (ϵ) was determined as a function of the mission energy requirements factor $\left(\frac{\Delta V}{c^x}\right)$ and engine parameter factor $\frac{\eta}{P(1+k)}$. This is plotted in Fig. 3-16, 3-17, and 3-18 for various nozzle weight factors (ρt).

Initial Thrust-to-Earth Weight Ratio

Consider Eq. (1) and (2) and the following:

1. k, f, c^x, C_F = constants
2. ΔV = function of η
3. Maximize μ

Take derivatives of Eq. (1) and (2) with respect to η and let $\frac{\partial \mu}{\partial \eta} = 0$.

$$\frac{e^{-\frac{\Delta V}{c^x C_F}}}{c^x C_F} \; \frac{\partial \left[\Delta V\right]}{\partial \eta} = - \frac{f}{1 + k}$$

Let $c^x C_F = gI$ where

I = specific impulse ~ sec

g = 32.2 ft/sec^2

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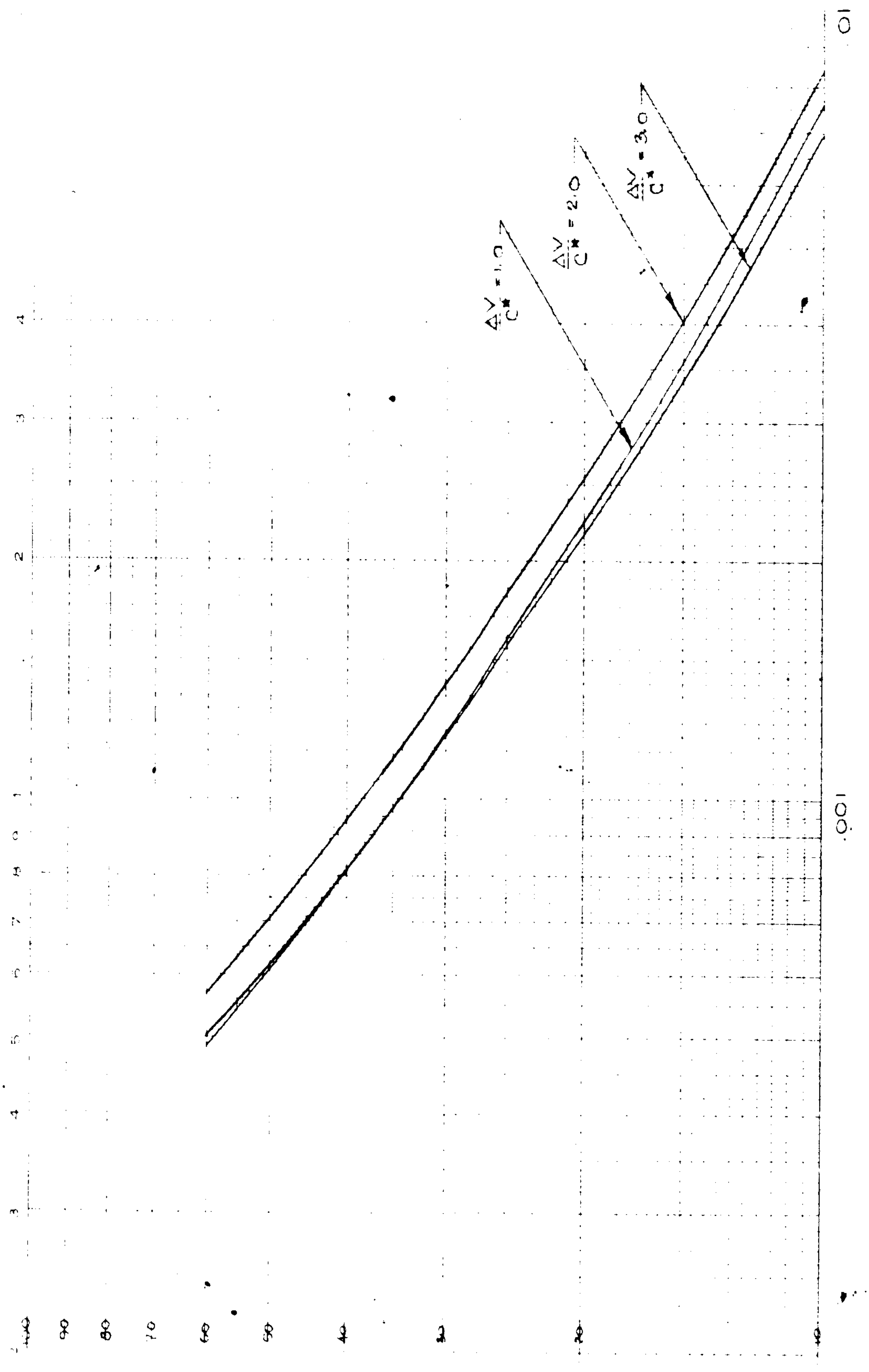

Figure 3-16. Optimum Expansion for Vacuum Operation Nozzle Specific Weight, ρ_t=0.2 lb/in.^{2}

[redacted]

[redacted]

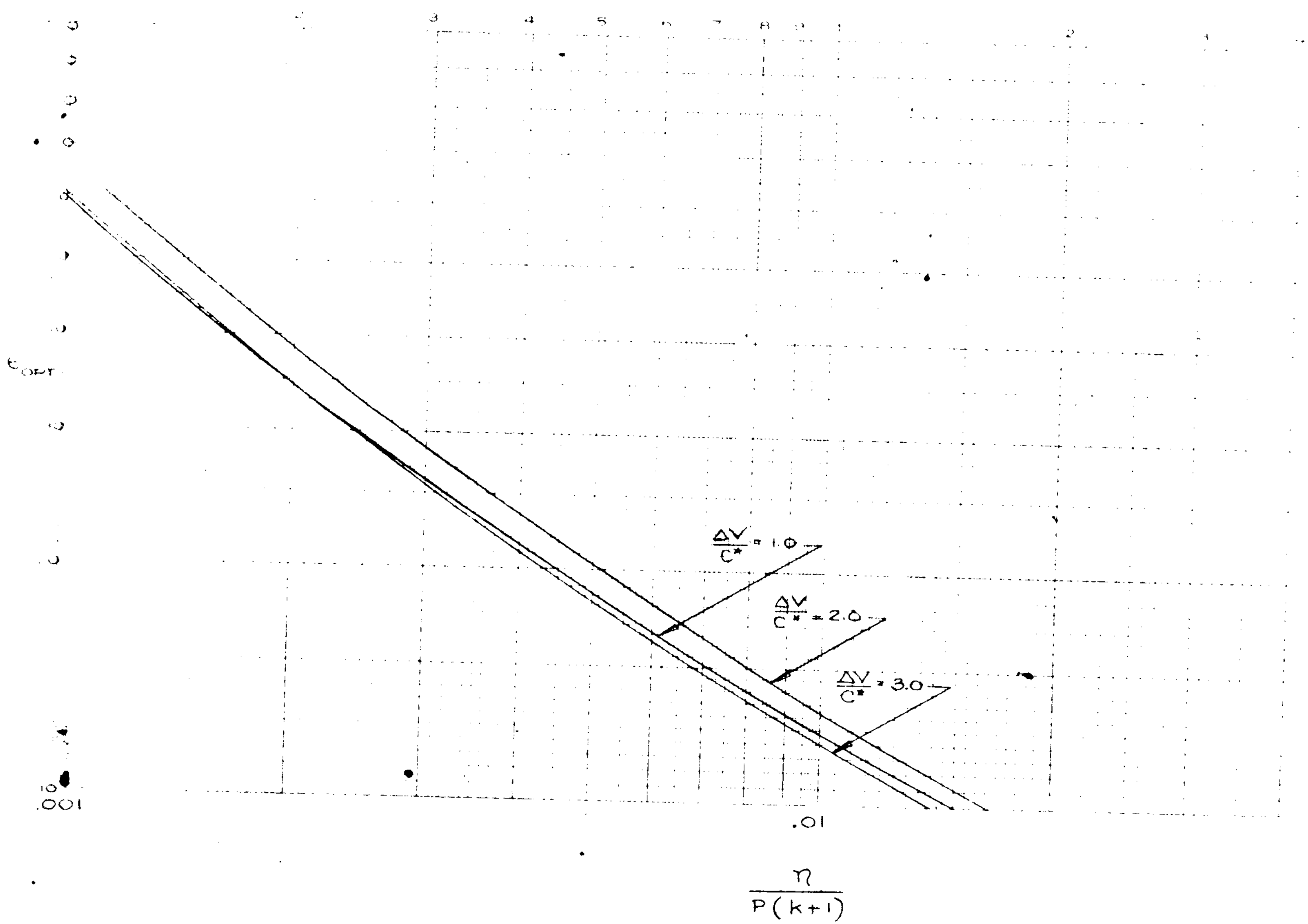

Figure 3-17. OPTIMUM EXPANSION FOR VACUUM OPERATION NOZZLE SPECIFIC WEIGHT, $\rho = 0.1$ lb/in^2

[redacted]

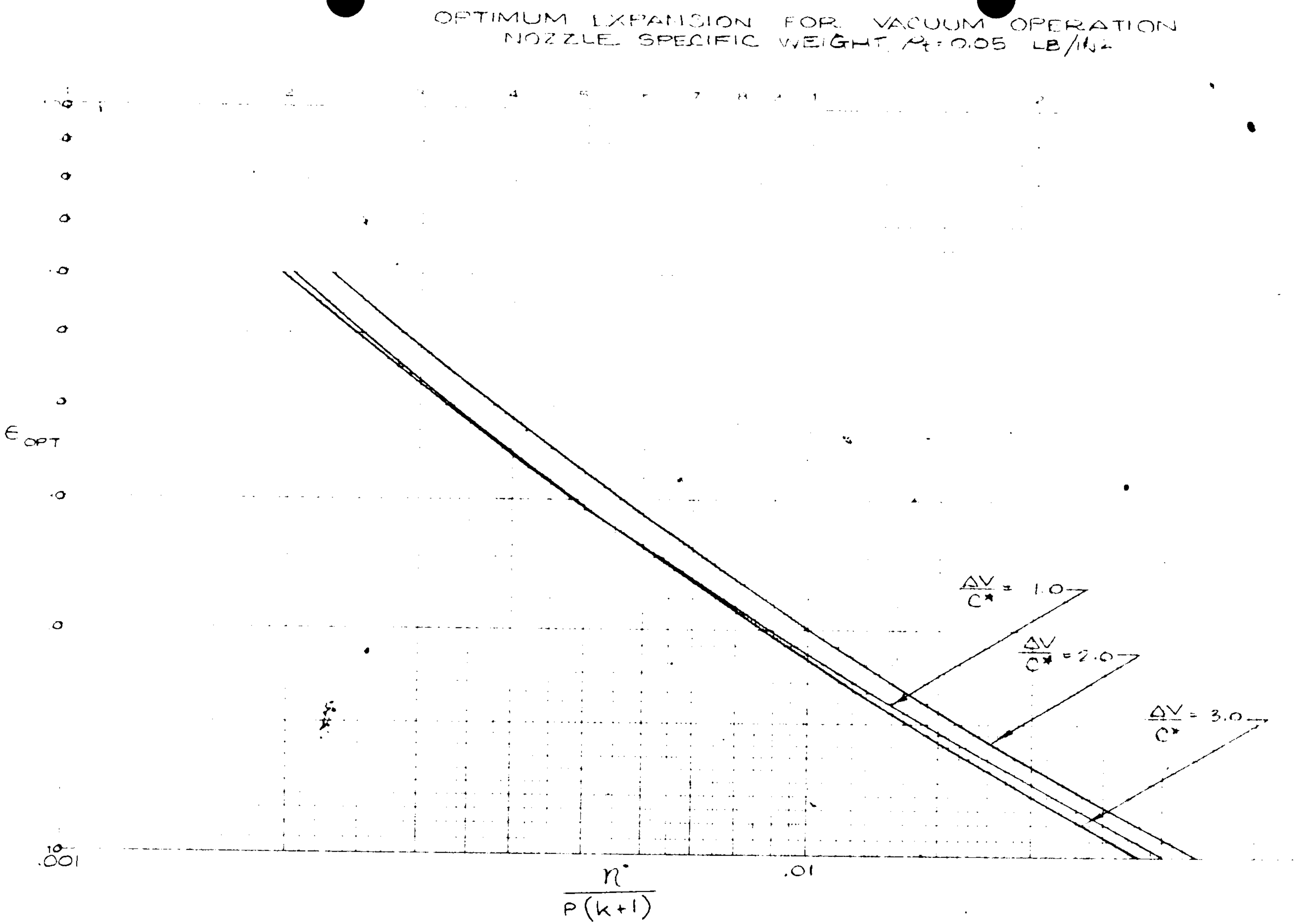

Figure 3-18. Optimum Expansion for Vacuum Operation Nozzle Specific Weight, ρ_t = 0.05 lb/in.^{2}

$$\frac{e^{-\frac{\Delta V}{gI}}}{g\,I}\;\frac{\partial\left[\Delta V\right]}{\partial \eta} = -\frac{f}{1+k}$$

For a given propulsive maneuver, ΔV and $\partial\left[\Delta V\right]/\partial \eta$ can be determined as functions of η. For five manuevers that might be used in a lunar mission, the optimum initial thrust-to-earth weight ratio was calculated as a function of the weight factor $\left(\frac{f}{1+k}\right)$ for an I of 420 sec. This is plotted in Fig.3-19. The missions considered were:

1. 2.6 day earth-moon transfer from 300 n mi earth orbit
2. 50 n mi lunar orbit establishment from 2.6 day transfer
3. Direct lunar landing from 2.6 day transfer; thrust parallel to velocity
4. Lunar landing from 50 n mi lunar orbit; ICP trajectory
5. Direct lunar takeoff for 2.6 day moon-earth transfer

Optimum Chamber Pressure

Considering again Eq.(1) and (2), the following assumptions were made

1. η, ΔV, C_F, constant
2. c^x, f, k = functions of P
3. μ to be maximum

Relations were developed for the three factors which are functions of chamber pressure (P).

[redacted]

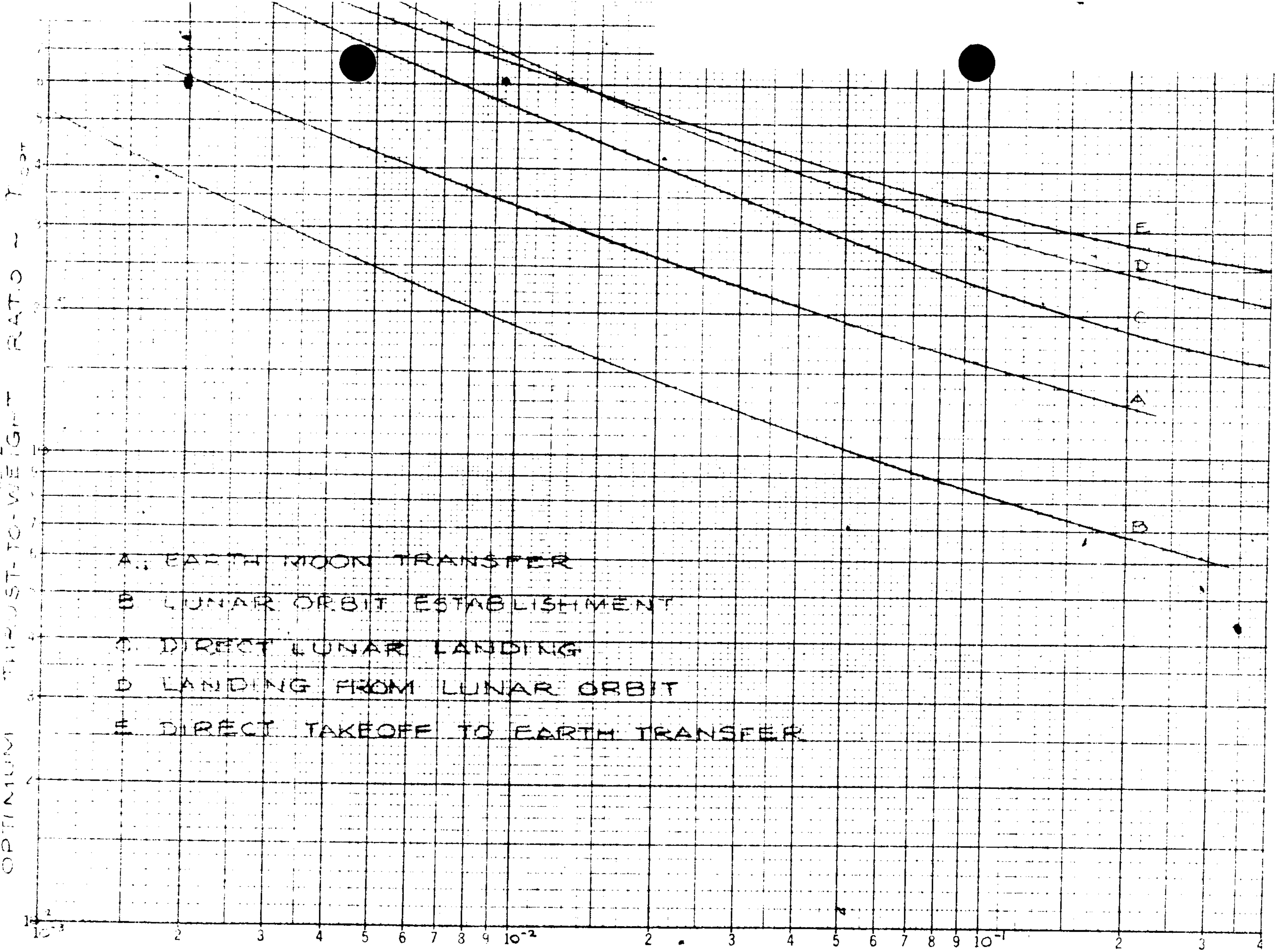

Figure 3-19. Optimum Initial Thrust-to-Weight Ratio for Lunar Mission Maneuvers $\cdot \left(\frac{f}{1+k}\right)$

[redacted]

$$c^x = \alpha \ln P + \beta$$

$$k = \frac{\gamma}{\rho B} P_T$$

$$f = \nu P_D + \frac{\phi}{P} + \psi$$

where

β , α = characteristic velocity constants

γ = structure constant ~ $\dfrac{\text{lb structure}}{\text{cu ft propellant - psi pressure}}$

P_T = tank pressure ~ psi

P_D = pump discharge pressure ~ psi

ρB = propellant bulk density ~ lb/cu ft

ν = turbopump specific weight constant ~ $\dfrac{\text{lb}}{\text{lb-psi}}$

ϕ, ψ = thrust chamber specific weight constants

Derivatives were taken and the equations were solved to give maximum μ . (Appendix B). The following equations resulted.

<u>Pump-Fed Systems</u>.

$$\left[\eta \nu \left(\frac{\partial P_D}{\partial P}\right)\right] P_{opt} = e^{-\frac{\Delta V}{g'_o}} \left[\frac{\Delta V}{g'_o}\right] \left[\frac{\alpha}{2c_o^x}\right] + \sqrt{\left\{ e^{-\frac{\Delta V}{g'_o}} \cdot \left[\frac{\Delta V}{g'_o}\right] \left[\frac{\alpha}{2c_o^x}\right] \right\}^2 + \eta^2 \phi \nu \left[\frac{\partial P_D}{\partial P}\right]}$$

The thrust chamber weight constant ϕ is a factor related to the tube bundle wet weight and does not include the effects of bands, injector, manifolds, etc.

Pressure-Fed Systems.

$$\left[\left(\frac{\gamma}{\rho_B}\right)\left(\frac{\partial P_T}{\partial P}\right)\right] P_{opt} = \frac{\left[\frac{\Delta V}{gI_o}\right]\left[\frac{\alpha}{2c_o^x}\right]}{\left[e^{\frac{\Delta V}{gI_o}} - 1\right]} + \sqrt{\left\{\frac{\left[\frac{\Delta V}{gI_o}\right]\left[\frac{\alpha}{2c_o^x}\right]}{\left[e^{\frac{\Delta V}{gI_o}} - 1\right]}\right\}^2 + \frac{\eta\phi}{\left[1 - e^{-\frac{\Delta V}{gI_o}}\right]}\,\frac{\gamma}{\rho_B}\,\frac{\partial P_T}{\partial P}}$$

In the derivation and subsequent plotting of the equations two simplifying assumptions were made. One, the variation in c* (and thesefore I) with chamber pressure can be neglected in the equations for P_{opt}. Two, the value of α/c^x depends upon the propellant combination being considered. To facilitate the plotting of the two equations and allow the effects of other parameters to be observed, a value of 0.00423 was assumed for $\alpha/2c^x$. This represents a more or less average value for a number of different propellant combinations. The optimum chamber pressure for pump-fed and pressure-fed systems are presented in Fig. 3-20 and 3-21 respectively.

[redacted]

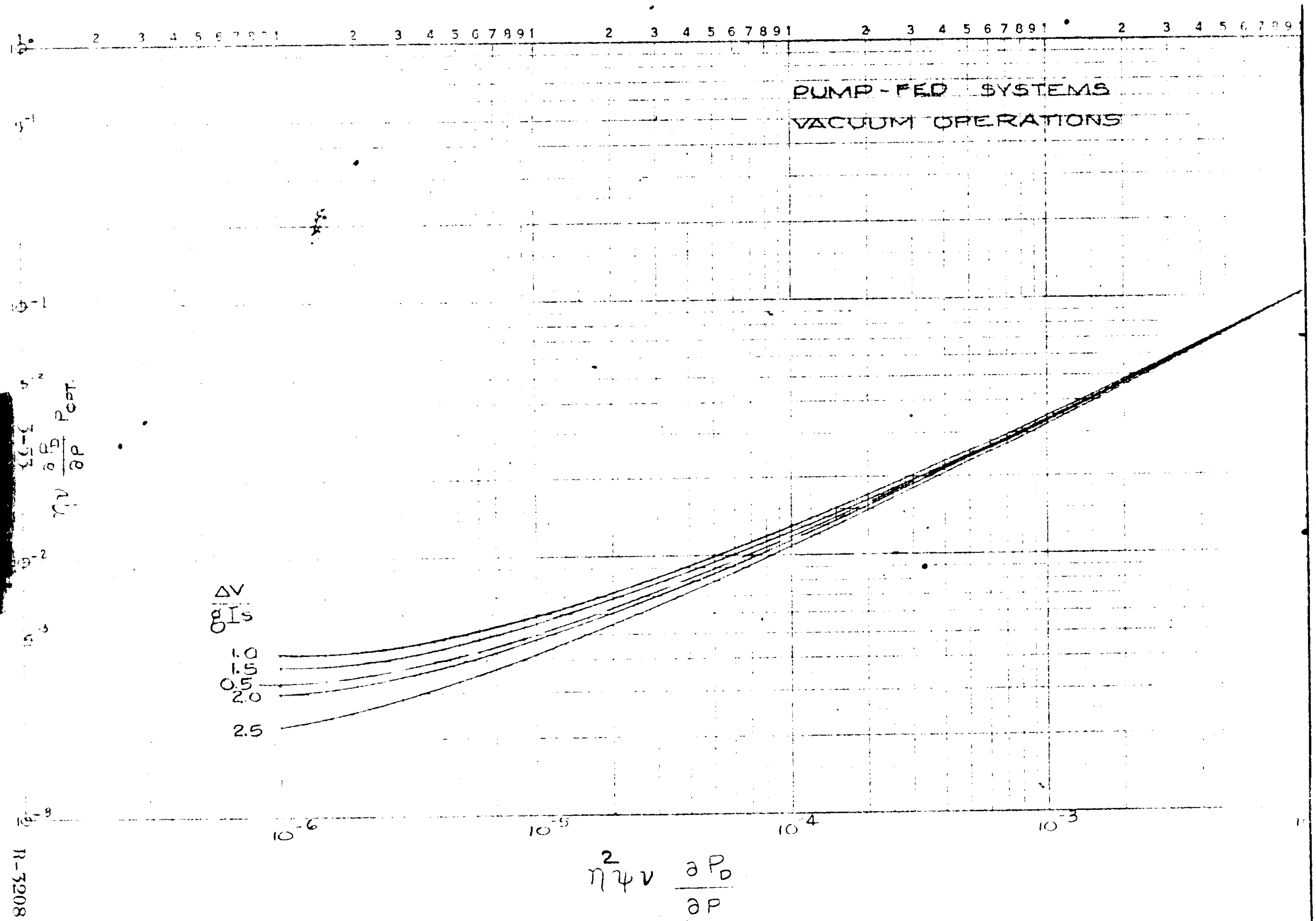

Figure 3-20. Chamber Pressure Optimization

[redacted]

[redacted]

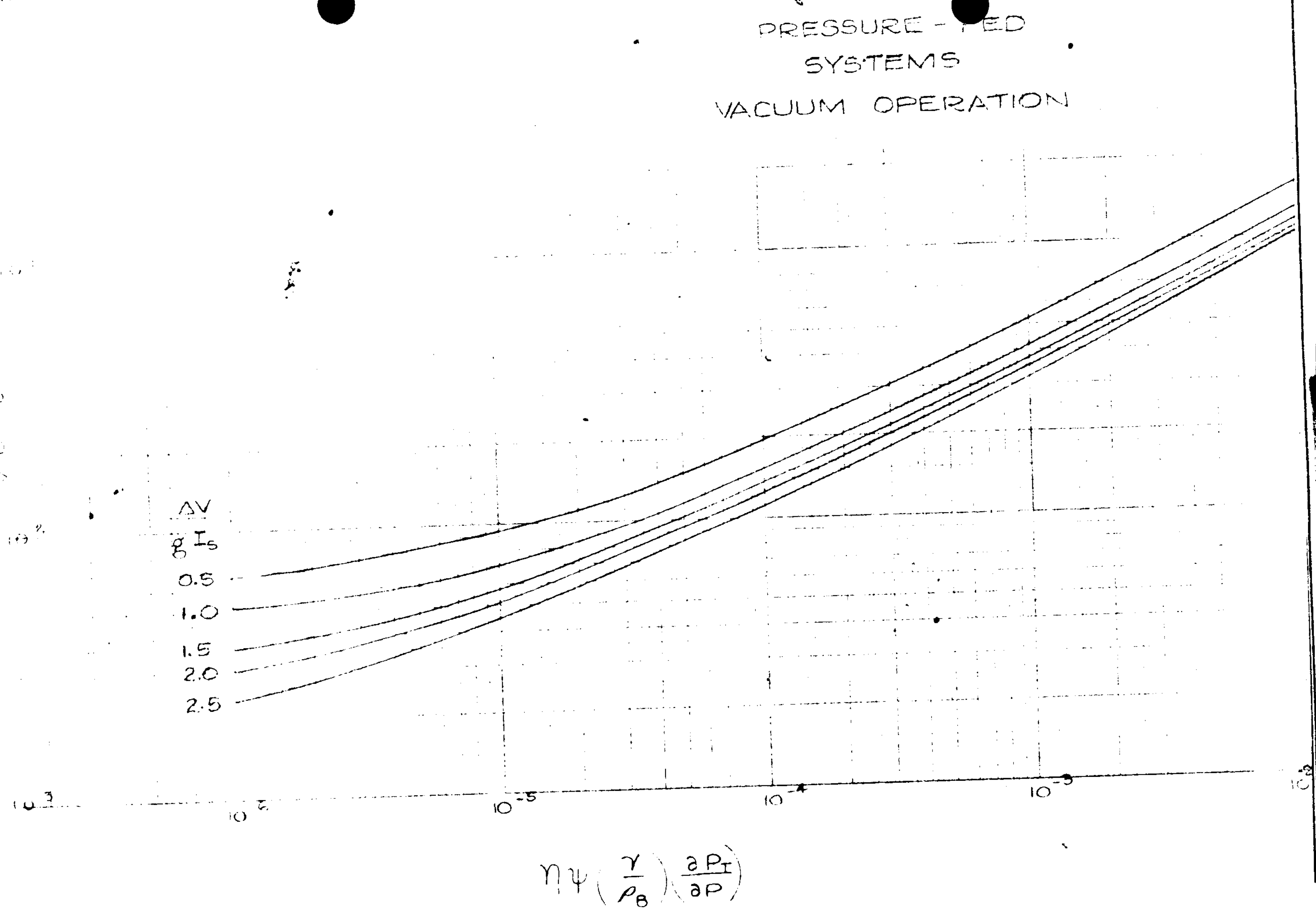

Figure 3-21. Pressure-Fed Systems Vacuum Operation

[redacted]

Engine Operating Parameter Tradeoff

To provide an indication of the influence of various factors on the value of these optimum parameters a brief look was taken at the parameter tradeoffs for a typical propulsion system. The values for the nominal points used in the different tradeoffs are listed below.

Thrust-to-Weight Ratio.

1. Earth/Moon transfer
2. Specific engine weight, $f = 0.015$ lb/lb
3. Specific tank weight, $k = 0.05$ lb/lb

Expansion Area Ratio.

1. Nozzle specific weight, $\rho t = 0.1$ lb/sq in.
2. Thrust-to-weight ratio, $\eta = 0.5$
3. Chamber pressure, $P = 150$ psi
4. Specific tank weight, $k = 0.05$ lb/lb
5. Ideal velocity increment, $\Delta V = 10\ 000$ fps
6. Characteristic velocity, $c = 5.000$ fps

Chamber Pressure

1. $\dfrac{\partial P_D}{\partial P} = \dfrac{\partial P_T}{\partial P} = 1.5$
2. Thrust-to-weight ratio, $\eta = 0.5$
3. Pump weight factor, $\nu = 0.00001$ lb/lb-psi
4. Thrust chamber weight factor, $\phi = 1.52 \dfrac{\text{lb-psi}}{\text{lb}}$
5. Ideal velocity increment, $\Delta V = 10{,}000$ fps
6. Specific impulse, $I_s = 312$ sec
7. Tank weight factor $\gamma = 0.015 \dfrac{\text{lb}}{\text{psi-ft}^3}$
8. Bulk density, $\rho_B = 50$ lb/cu ft

Using the figures and methods developed in the previous section, the significant factors involved in establishing the value of an engine operating parameter are varied ±50 percent and the effect on the parameter is as indicated in Fig.3-22 through 3-25.

The variation in the value of optimum thrust-to-weight ratio is shown in Fig.3-22 for a particular space mission. It can be seen that the tank specific weight (k) has negligible effect on the optimum. The engine specific weight (f) has a somewhat greater effect. As expected an increase in engine specific weight would cause the optimum to occur at a lower value of thrust-to-weight ratio. The greatest factor in causing changes in optimum thrust-to-weight ratio is the mission for which the parameter is being optimized.

The optimum expansion ratio variation is shown in Fig.3-23. Specific tank weight (k) has negligible effect on the optimum value. The other factors: thrust-to-weight (η), chamber pressure (P), and specific nozzle weight (ρt) affect the optimum expansion ratio through variation in thrust chamber weight. As the factor acts to increase thrust chamber weight the optimum expansion ratio value is decreased and visa versa.

Optimum chamber pressure variation is shown in Fig.3-24 and 3-25 for pressure-fed and pump-fed systems respectively. The effect of the various factors on the optimum value is similar for the two systems with an increase in factors increasing thrust chamber weight tending toward higher optimum chamber pressures, and increases in factors increasing feed system weight tending toward lower optimum values. It is interesting to note the effect of thrust-to-weight ratio which acts in an opposite manner for the two systems. In the pressure-fed system an increase in thrust-to-weight ratio causes the chamber weight to increase relative to the feed system. The optimum value of chamber pressure increases to compensate for this increased thrust chamber weight. In the pump-fed system the thrust-to-weight ratio affects the chamber pressure in an opposite manner. An increase in thrust-to-weight ratio causes increases in both thrust chamber and feed system weights. For this particular case the feed system weight variation is more significantly affected by thrust-to-weight ratio than the thrust chamber weight variation. The optimum chamber pressure shifts to compensate for this relative weight increase.

[redacted]

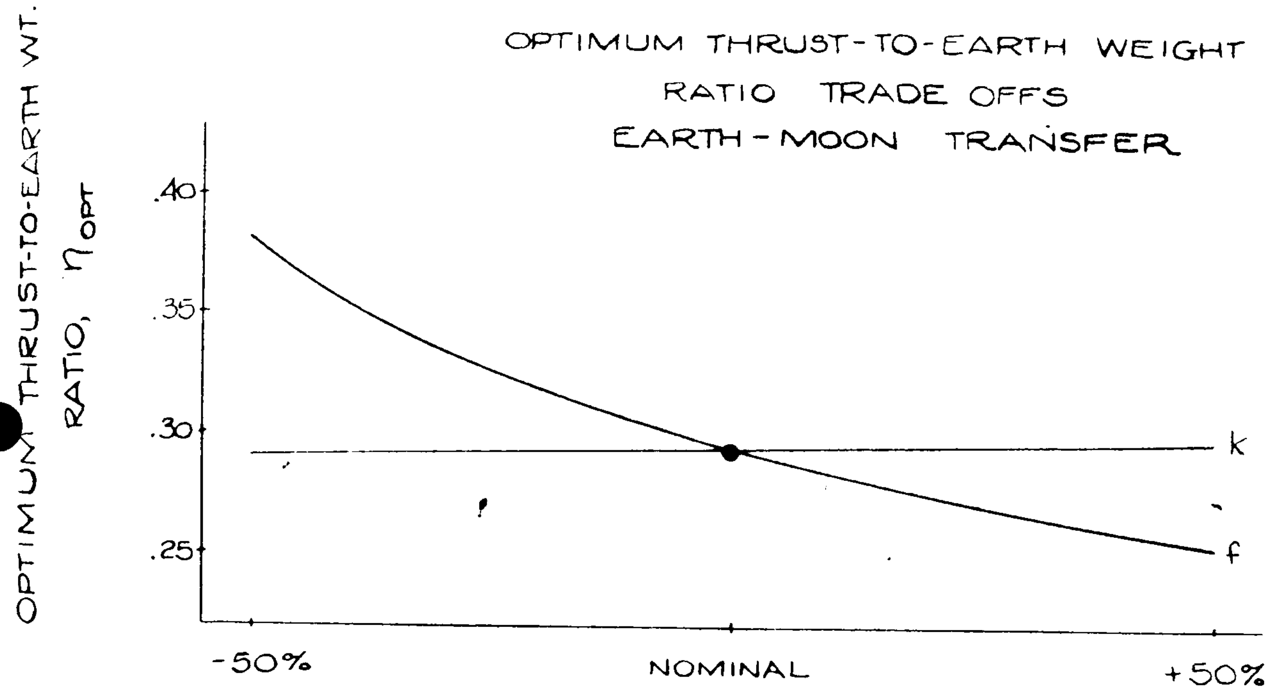

Figure 3-22. FACTOR VARIATION

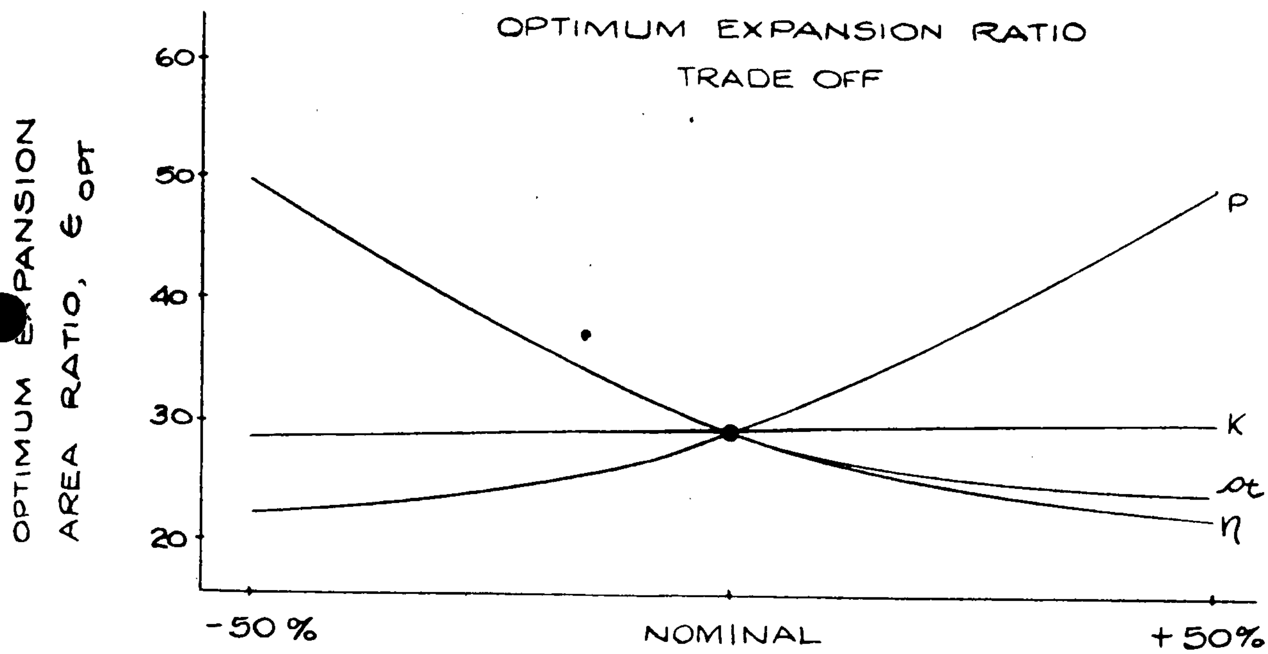

Figure 3-23. FACTOR VARIATION

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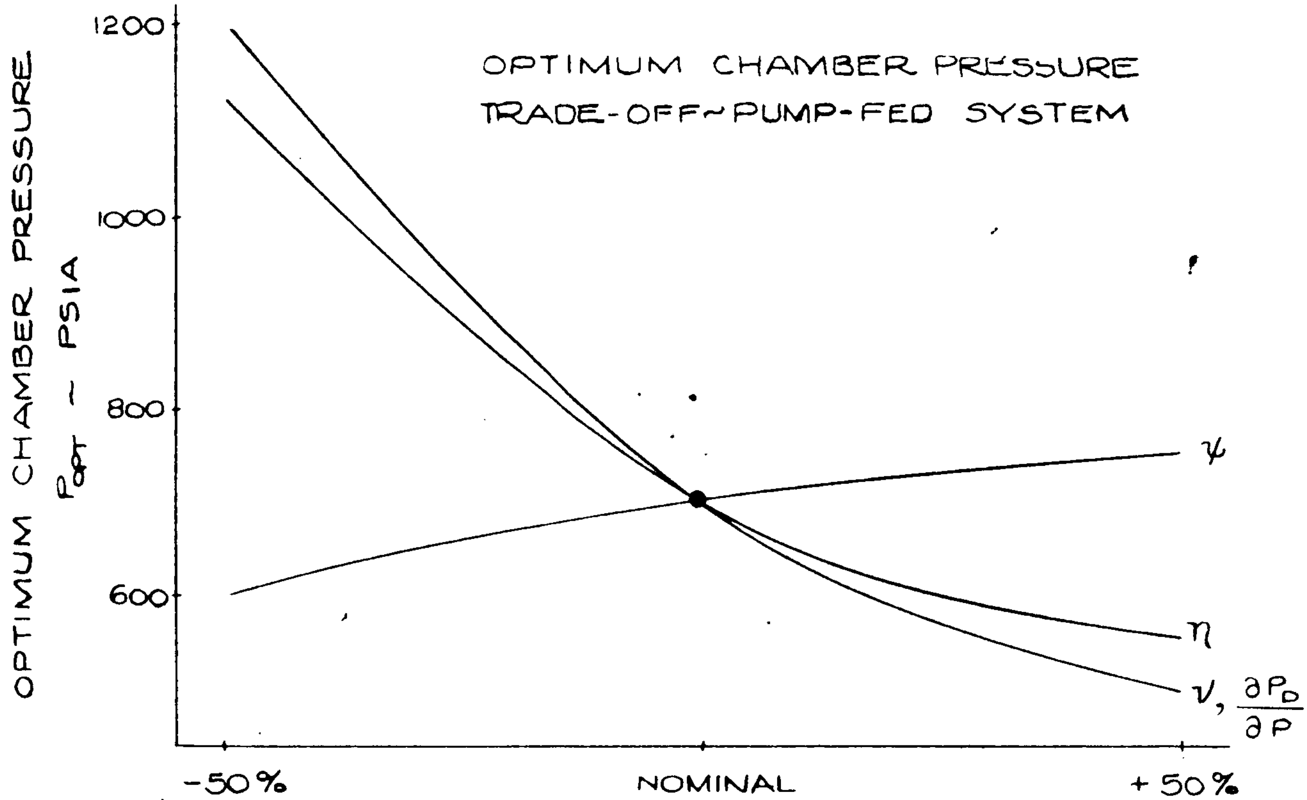

Figure 3-24 . FACTOR VARIATION

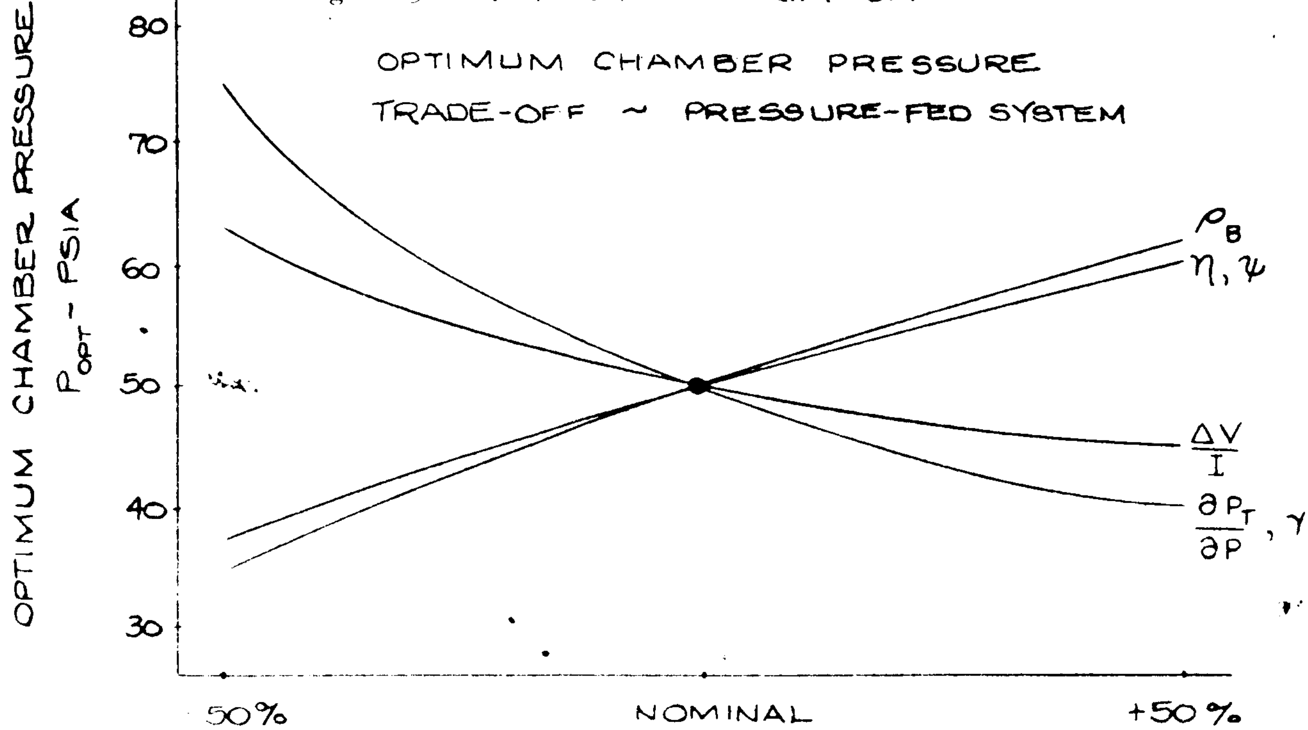

Figure 3-25. FACTOR VARIATION

[redacted]

[redacted]

APPENDIX A

OPTIMUM EXPANSION RATIO

For the optimum expansion ratio determination the following assumptions were made

1. c^*, k, η, V = constant
2. f, C_F = functions of
3. μ to be maximum

Substitute Eq.(1) in Eq.(2), take derivatives with respect to ϵ, and set $\partial\mu/\partial\epsilon$ equal to zero,

$$\left[\frac{\Delta V}{c^x C_F}\right] e^{-\frac{\Delta V}{c^x C_F}} \left[\frac{1}{C_F}\left(\frac{\partial C_F}{\partial \epsilon}\right)\right] = \left(\frac{\eta}{1+k}\right)\left(\frac{\partial f}{\partial \epsilon}\right) \qquad (3)$$

For a conical nozzle the weight factor f can be expressed as

$$f = \text{constant} + \frac{\rho t A}{F \sin \alpha}\left(\epsilon - \epsilon_o\right)$$

where

ρt = nozzle weight factor, lb/sq in.

A = throat area, sq in.

α = nozzle half-angle

[redacted]

[redacted]

F = thrust, lb = PAC_F

ϵ_o = initial expansion ratio

Writing A/F as $1/PC_F$ and taking derivatives with respect to ϵ

$$\frac{\partial f}{\partial \epsilon} = \frac{\rho t}{PC_F \sin \alpha} \left[1 - (\epsilon - \epsilon_o) \left(\frac{1}{C_F} \frac{\partial C_F}{\partial \epsilon}\right)\right]$$

$$(\epsilon - \epsilon_o) \left(\frac{1}{C_F} \frac{\partial C_F}{\partial \epsilon}\right)$$

is fairly small with respect to 1 and was considered negligible. Therefore

$$\frac{\partial f}{\partial \epsilon} = \frac{\rho t}{PC_F \sin \alpha}$$

Substituting this in Eq. [illegible].

$$\left[\frac{\Delta V}{c^x}\right] e^{-\left(\frac{\Delta V}{c^x}\right)\left(\frac{1}{C_F}\right)} \left[\frac{1}{C_F} \left(\frac{\partial C_F}{\partial \epsilon}\right)\right] = \left(\frac{\eta}{1 + k}\right) \frac{\rho t}{P \sin \alpha}$$

[redacted]

APPENDIX B

OPTIMUM CHAMBER PRESSURE

Considering Eq.(1) and(2) the following assumptions were made

$$\eta,\ \triangle V,\ C_F = \text{constant}$$

$$c^x = \alpha \ \ln P + \beta \ ; \quad \frac{\partial c^x}{\partial P} = \frac{\alpha}{P}$$

$$k = \frac{\gamma}{\rho_B} P_T ; \quad \frac{\partial k}{\partial P} = \frac{\gamma}{\rho_B} \left(\frac{\partial P_T}{\partial P} \right)$$

$$f = v\ P_D + \frac{\phi}{P} + \psi \ ; \quad \frac{\partial f}{\partial P} = v \left(\frac{\partial P_D}{\partial P} \right) - \frac{\phi}{P^2}$$

where

β, α = characteristic velocity constants

γ = structure constant $\sim \dfrac{\text{lb structure}}{\text{cu ft propellant - psi pressure}}$

P_T = tank pressure ~ psi

P_D = pump discharge pressure ~ psi

ρ_B = propellant bulk density ~ lb/cu ft

ν = specific weight constant of turbopump $\sim \dfrac{\text{lb pump}}{\text{lb-propellant-psi pressure}}$

ϕ, ψ = thrust chamber specific weight constants

[redacted]

Take derivatives of Eq. 1 and 2 with respect to P, and set $\frac{\partial \mu}{\partial P} = 0$.

$$\frac{\partial k}{\partial P} + \eta \frac{\partial f}{\partial P} - \frac{1}{R}\left[\left(\frac{\partial k}{\partial P}\right) - \frac{1+k}{R}\left(\frac{\partial R}{\partial P}\right)\right] = 0$$

$$\frac{\partial R}{\partial P} = -\left[\frac{\Delta V}{{c^x}^2 C_F}\right] e^{\frac{\Delta V}{c^x C_F}} \left(\frac{\partial c^x}{\partial P}\right)$$

Since $k \ll 1$ we can write, combining the equations

$$\frac{\partial k}{\partial P} + \eta \frac{\partial f}{\partial P} - e^{-\frac{\Delta V}{c^x C_F}} \left[\left(\frac{\partial k}{\partial P}\right) + \left(\frac{\Delta V}{{c^x}^2 C_F}\right)\left(\frac{\partial c^x}{\partial P}\right)\right] = 0$$

Substituting the assumed relations

$$\left(\frac{\gamma}{\rho_B}\right)\left(\frac{\partial P_T}{\partial P}\right) + \eta \nu \left(\frac{\partial P_D}{\partial P}\right) - \frac{\eta \phi}{P_{opt}^2}$$

$$= e^{-\frac{\Delta V}{c^x C_F}} \left[\left(\frac{\gamma}{\rho_B}\right)\left(\frac{\partial P_T}{\partial P}\right) + \left(\frac{\Delta V}{{c^x}^2 C_F}\right)\left(\frac{\alpha}{P_{opt}}\right)\right]$$

[redacted]

Assuming that variations in $\dot{c}^*$ can be neglected in this equation and rearranging

$$\left\{ \left(\frac{\gamma}{\rho_B}\right) \left[\frac{\partial P_T}{\partial P}\right] \left[1 - e^{-\frac{\Delta V}{c_o^x C_F}}\right] + \eta \nu \left(\frac{\partial P_D}{\partial P}\right) \right\} P_{opt}^2 -$$

$$\left\{ e^{-\frac{\Delta V}{c_o^x C_F}} \left[\frac{\Delta V}{c_o^x C_F}\right] \frac{\alpha}{c_o^x} \right\} P_{opt} - \eta \phi = 0$$

Equations for the pump-fed system are as follows:

$$\frac{\partial P_T}{\partial P} = 0$$

$$P_{opt}^2 - \left\{ \frac{e^{-\frac{\Delta V}{c_o^x C_F}}}{\eta \nu \left(\frac{\partial P_D}{\partial P}\right)} \left[\frac{\Delta V}{c_o^x C_F}\right] \left(\frac{\alpha}{c_o^x}\right) \right\} P_{opt} - \frac{\phi}{\nu} \left[\frac{1}{(\partial P_D / \partial P)}\right] = 0$$

$$\left[\eta \nu \left(\frac{\partial P_D}{\partial P}\right)\right] P_{opt} = e^{-\frac{\Delta V}{c_o^x C_F}} \left[\frac{\Delta V}{c_o^x C_F}\right] \left(\frac{\alpha}{2 c_o^x}\right) +$$

$$\sqrt{\left\{ e^{-\frac{\Delta V}{c_o^x C_F}} \left[\frac{\Delta V}{c_o^x C_F}\right] \frac{\alpha}{2 c_o^x} \right\}^2 + \eta^2 \phi \nu \left(\frac{\partial P_D}{\partial P}\right)}$$

[redacted]

Equations for the pressure-fed system are as follows:

$$\frac{\partial P_D}{\partial P} = 0$$

$$\left\{ \left(\frac{\gamma}{\rho_B}\right) \left(\frac{\partial P_T}{\partial P}\right) P_{opt} \right\}^2 \left[1 - e^{-\frac{\Delta V}{c_o^x C_F}} \right] - \left\{ e^{-\frac{\Delta V}{c_o^x C_F}} \left[\frac{\Delta V}{c_o^x C_F}\right] \frac{\alpha}{c_o^x} \right\} \left[\left(\frac{\gamma}{\rho_B}\right) \left(\frac{\partial P_T}{\partial P}\right) P_{opt} \right] - \eta \phi \left(\frac{\gamma}{\rho_B}\right) \left(\frac{\partial P_T}{\partial P}\right) = 0$$

$$\left(\frac{\gamma}{\rho_B}\right) \left(\frac{\partial P_T}{\partial P}\right) P_{opt} = \frac{\left[\frac{\Delta V}{c_o^x C_F}\right] \left[\frac{\alpha}{2 c_o^x}\right]}{\left[e^{\frac{\Delta V}{c_o^x C_F}} - 1 \right]} + \sqrt{ \left\{ \frac{\left[\frac{\Delta V}{c_o^x C_F}\right] \left[\frac{\alpha}{2 c_o^x}\right]}{\left[e^{\frac{\Delta V}{c_o^x C_F}} - 1 \right]} \right\}^2 + \frac{\eta \phi}{\left[1 - e^{-\frac{V}{c_o^x C_F}} \right]} \left(\frac{\gamma}{\rho_B}\right) \left(\frac{\partial P_T}{\partial P}\right) }$$

[redacted]

[redacted]

SPACE ENVIRONMENT EFFECTS ON PROPULSION SYSTEMS

Summary

The significant environmental problems to be encountered in space are those of meteoroid penetration, heat exchange (thermal radiation), hard vacuum, particulate radiation, and zero gravity. In spite of the attention these conditions have had in the literature, most of the work is theoretical and general. The succeeding sections review the environmental conditions and their effects on the propulsion system. Those conditions that are most significant in their effects on the propulsion system are summarized along with a brief discussion of design procedures of propulsion system protection.

Thermal radiation, meteoroids, and zero gravity are the most significant environmental conditions as far as the propulsion system is concerned. The remaining environmental conditions, radiation and hard vacuum, affect the propulsion system largely from a materials standpoint. Through proper material selection these problems can be eliminated.

Studies of the thermal radiation problem indicate the significant effect of this environment on the propulsion system particularly in the matter of propellant selection. Figures 3-32 through 3-34 indicate that for long storage times the insulation necessary to store liquid hydrogen may make storable propellant combinations more attractive.

Applying the general results of these studies to the missions and vehicles presently being considered, two important effects can be observed. One, storage duration for all of the present missions is less than a year. Two, the main propulsion system propellant weights in the present vehicles are fairly large. Both of these effects tend to relieve the problems of storage. Considering Fig. 3-34 it is, therefore, unlikely

[redacted]

that propellant storage considerations will significantly alter the relative performance of the various propellant combinations, and that the high-energy propellants will provide maximum payloads. A positive decision on these effects would necessitate a fairly detailed study of the particular mission and vehicle. Conduction heat transfer between the various portions of the vehicle should be considered. For vehicles smaller than those presently under study the storage problem may become significant, particularly for the second phase of the Mars mission.

For the lunar mission storage times of five to eight days may be required. Figure 3-34 indicates that a very slight amount of insulation would be necessary for "no loss" storage. To be conservative, an insulation thickness of 3/4 in. was assumed for cryogenic propellants. For the Linde SI-4, this resulted in a specific weight of about 0.3 lb/sq ft of surface. Insulation requirements for "Earth" storable propellants were assumed negligible.

Mars missions (one-way) may require storage times on the order of 200 days. Using 3 in. of insulation, and allowing an increase in tank pressure, the hydrogen can be maintained with less than 10-percent loss for this duration. Proper attitude control, and larger pressure and ullage allowances would permit storage with negligible or no losses. An insulation thickness of 4 in. was conservatively assumed for hydrogen (1.50 lb/sq ft). Somewhat less insulation would be necessary for cryogenics such as oxygen or fluorine. Insulation requirements for storables were considered negligible.

Studies of the meteoroid problem have indicated that for the smaller, most frequent size of meteoroid, the Whipple meteor bumper is a very promising protection system. The chances of encountering larger meteoroid sizes increase with vehicle size and trip time. Therefore, the amount of protection will probably increase also. For the very large meteors direct protection is impossible, and some evasive or prediction system may be necessary. Based upon the studies made to date, a meteor bumper of 0.032-in. aluminum was assumed for the vehicles considered. It was felt that this shield, along with the insulation previously mentioned, would be adequate.

The zero gravity problem is common to all systems, and is primarily a design problem. Providing propellant for engine start in zero gravity may be accomplished through a number of design methods that are discussed in the latter portion of this section.

A review of various aspects of these space environment problems follows.

Thermal Radiation

Vehicles in space are undergoing a continuous heat exchange between the propulsion system and its external environment, and between the various portions of the propulsion system. Figure 3-26 indicates the heat flow paths of a typical propulsion system. In this figure it can be seen that internal heat exchange through conduction and internal radiation occurs, as well as external heat exchange between the system and its environment.

Conduction and Internal Radiation. The control of conduction and internal radiation is fairly straightforward. This internal heat transfer is largely a function of the design of the propellant tanks and their connecting structure. As illustrated by the figure, conduction will occur through

[redacted]

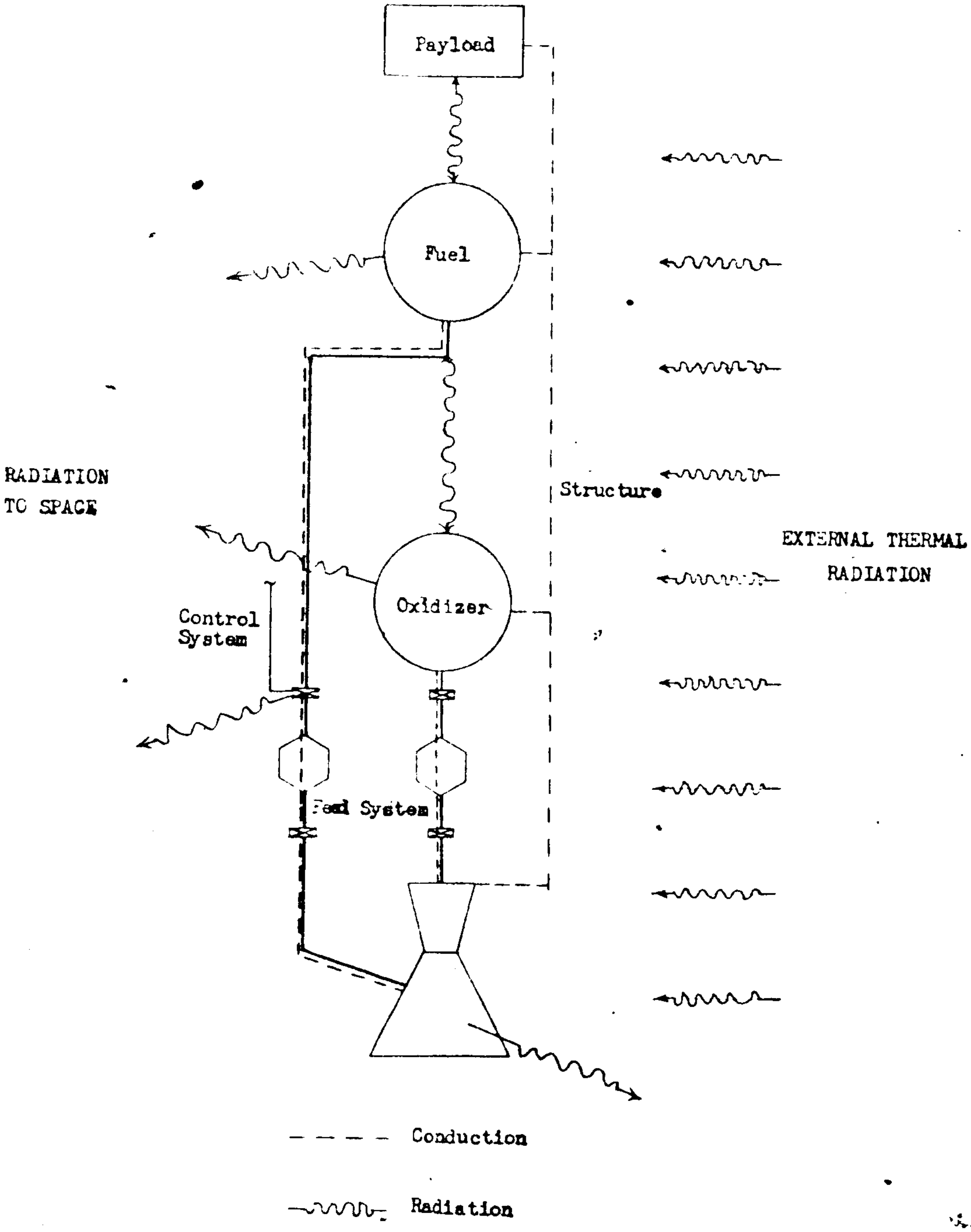

Figure 3-26. Heat Flow Paths of Typical Propulsion System

[redacted]

the feed lines and supporting structure. The internal radiation occurs between the various components, the payload, and the propellant tanks; it can be controlled by the selection of surface materials with proper emissivities and the insulation of the emitting surfaces.

Although seemingly insignificant, when viewed in terms of the long storage times required of the propulsion system on extended space missions, the conduction heat transfer between the propellant tanks may be the critical factor in limiting the storability of certain propellants. The inherent temperature differences between certain propellants (cryogenics) and the rest of the propulsion system call for a high degree of thermal insulation. This requirement for good insulation is the antithesis of the requirement for good structural support. The complications of designing a strong structural member that allows low heat conduction call for large amounts of design ingenuity. The effects of the conduction on the storability of cryogenics are illustrated in a later section.

<u>External Heat Exchange</u>. The external heat exchange consists of radiation input from the sun and any nearby planet, and radiation output from the propulsion system itself. The thermal radiation sources in the vicinity of a planet consist of solar radiation, planetary thermal radiation, and planetary reflected (albedo) radiation. The control of this external heat exchange can be affected by (1) control of the vehicle attitude with respect to the heat sources, (2) proper selection of surface conditions (absorptivity, emissivity), and (3) the use of insulation on emitting and absorbing surfaces. This control is illustrated by Fig. 3-27 and 3-28.

[redacted]

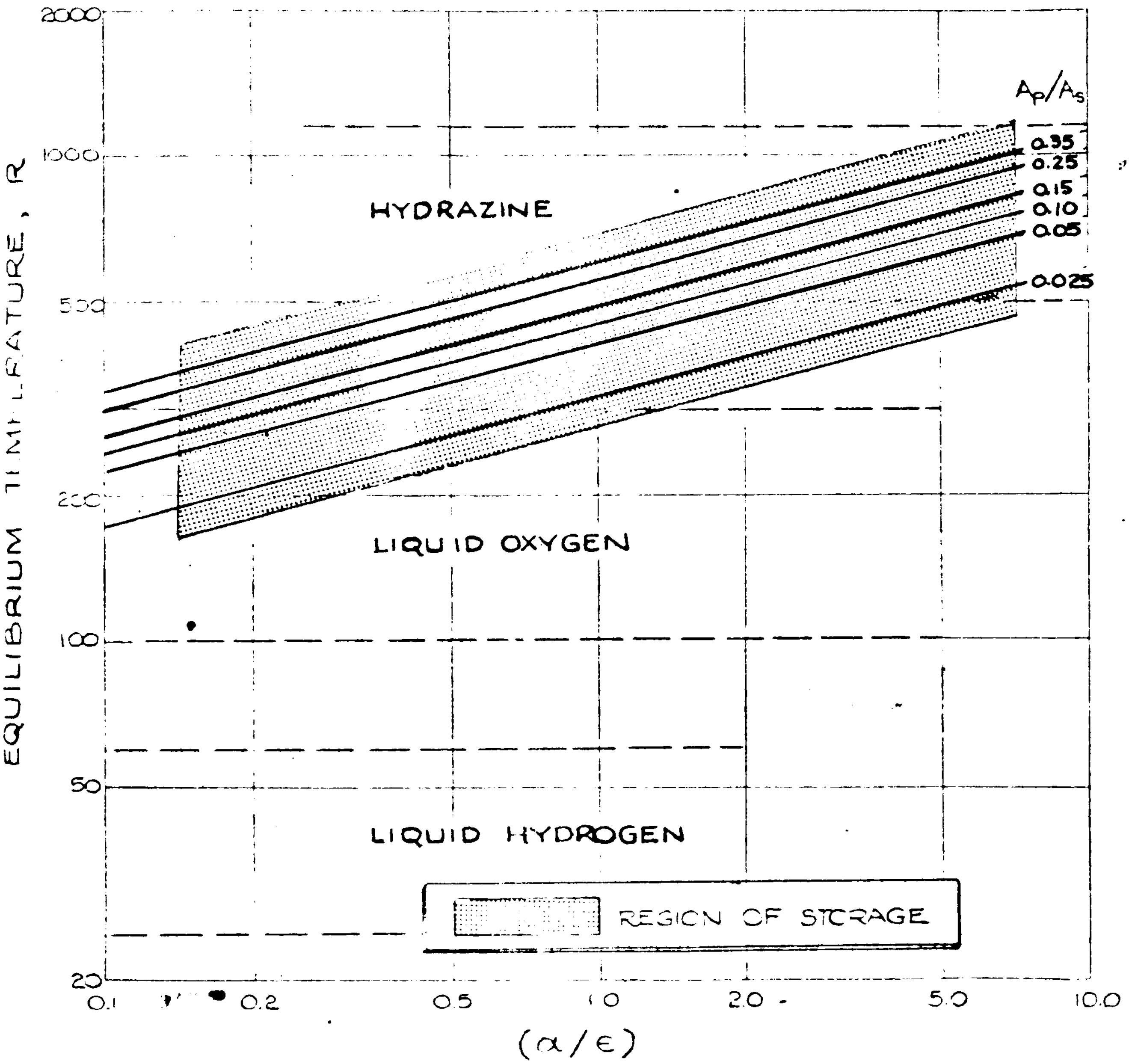

Figure 3-27. Equilibrium Temperature at 1 A.U. From Sun, Single Surface Material (Ref. 1)

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[redacted]

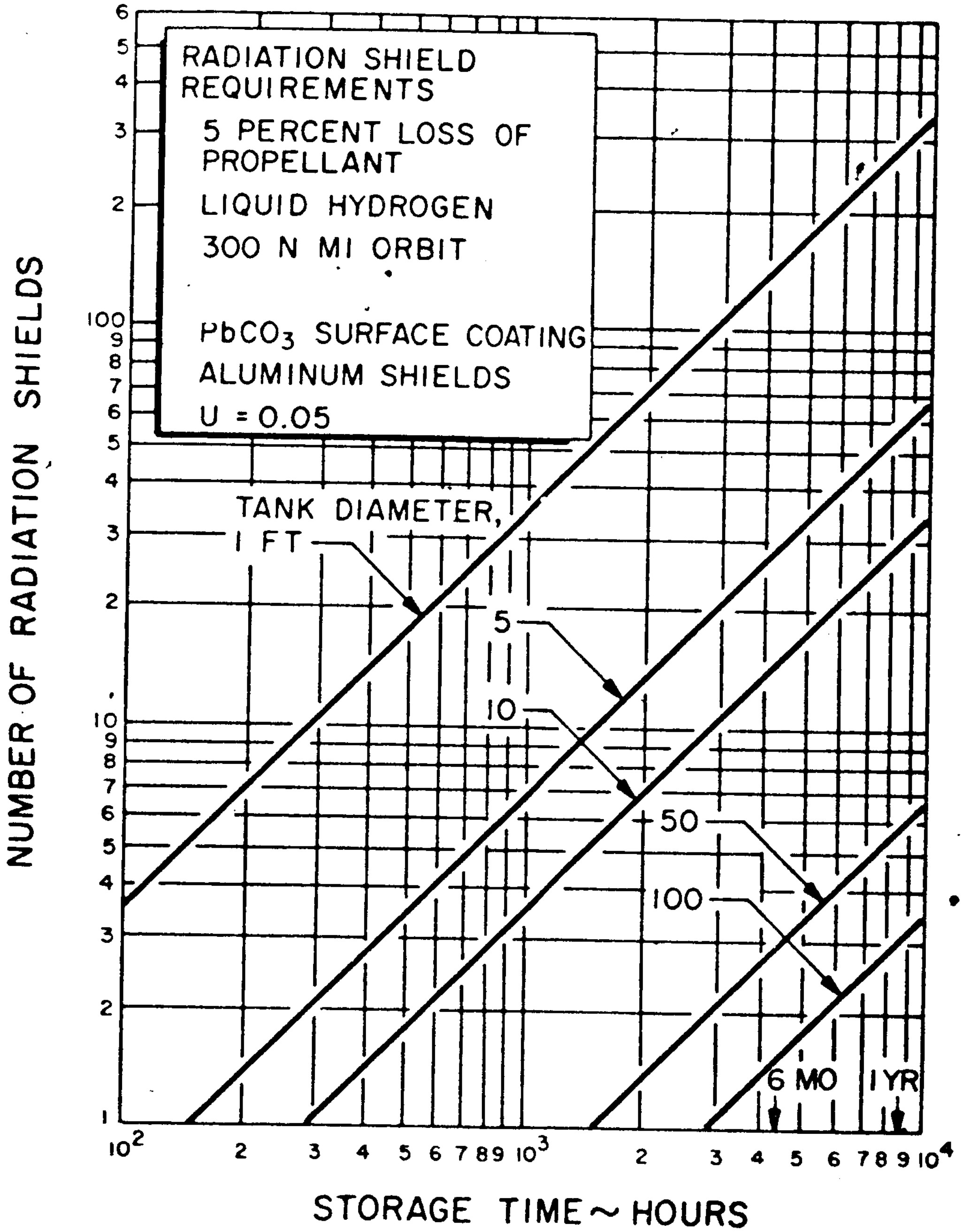

Figure 3-28. Number of Radiation Shields vs Storage Time (Ref. 2)

[redacted]

Figure 3-27 shows the effects of the ratio of projected (A_p) to surface (A_s) area (a function of attitude with respect to the radiation source) and the absorptivity to emissivity ratio on the equilibrium temperature of a propellant tank in space. This temperature corresponds to the situation where the heat leaving and entering the system are equal. The liquidus region of various propellant combinations are superimposed on the curves. The shaded region of the figure indicates the region in which the propellants can be easily stored. It can be seen that the hydrazine and oxygen are relatively easy to store compared to the hydrogen.

The storage of hydrogen cannot be accomplished on an equilibrium basis, i.e., with zero net heat flow. Insulation must be applied to the outside of the tank to regulate the heat influx and maintain the hydrogen temperature within the desired range over the time of storage. Figure 3-28 indicates the insulation requirements in terms of the number of radiation shields as a function of storage time and tank size. This figure assumes 5 percent of the hydrogen is evaporated during storage. It can be seen that the number of shields increase as storage time increases, and decrease as the size of the tank (amount of propellant) increases.

<u>Effects on Propulsion System</u>. A discussion of the effects of heat exchange in space on the propulsion system is found in a Rocketdyne study (Ref. 3). Successful operation of typical liquid propellant propulsion systems requires (1) that the propellants (other than hydrogen which is always at a critical temperature as it emerges from a thrust chamber regeneratively cooling jacket) be liquid in the combustion chamber injector passages, (2) that the propellant vapor pressures be low enough in turbopump-fed systems to allow provision of adequate

NPSH without exceeding tank pressure designs and (3) that the fuel be at low enough temperatures to adequately cool the thrust chamber. With this in mind, Ref. 3 assigns the following temperature limits to the propellants in the systems that were considered:

1. "The lower temperature limit in all cases is the freezing point."
2. "The upper temperature for hydrazine is 100 F to allow adequate chamber cooling."
3. "In the turbopump-fed systems, the upper temperature limit for each propellant except hydrazine is that which corresponds to a vapor pressure 20 psi below tank design pressure."
4. "In the pressure-fed systems, the upper temperature limit for each propellant except hydrazine is that which corresponds to a vapor pressure 5 psi less than the chamber pressure. The 5 psi is a margin of safety preventing vaporization in the injector."

These temperature limits apply to typical propulsion systems as described in the referenced report. In a final system design, a detailed study would be necessary to establish these temperature limits. If the propellant tanks are designed to accommodate the propellant volume increase accompanying rises in temperature, the above criteria would require no loss of propellant since the vapor pressure never reaches the tank design pressure.

[redacted]

Vehicle Performance Effects. From consideration of a number of studies of propellant storage (Ref. 1 through 6) it appears feasible to store any of the present propellants in any of the contemplated thermal radiation environments. This storage is accomplished through application of conductive and radiative insulation. Both cryogenic (H_2, etc.) and "Earth Storable" (N_2H_4, etc.) propellants can be stored for extended periods of time in the thermal environments of the Earth (Ref. 2), the moon (Ref. 4), and Mars (Ref. 5). The crux of the propellant storage problem is not so much whether a propellant can be maintained, but the cost of maintenance in terms of insulation weight.

From a study of liquid hydrogen storage (Ref. 6) Fig. 3-29 and 3-30 were obtained. These figures are for a specific tank design and assume that through proper attitude control and outer surface coatings, a mean outer skin temperature of 360 R can be attained. Linde SI-4 insulation is then placed between the outer and inner skins to vary the heat input to the propellant.

Figure 3-29 considers the evaporation rate of liquid hydrogen at its normal boiling point as a function of tank capacity and insulation thickness. Figure 3-30 considers "zero loss" hydrogen storage where the internal pressure is allowed to rise as the heat flows into the liquid. The curve on the right-hand side of the figure presents the limiting case of infinite insulation or zero heat input from outer tank surfaces. The heat flow into the propellant is entirely from conduction through piping and support structure. As tank capacity is increased, these losses become less significant and the "no loss" storage times can be increased.

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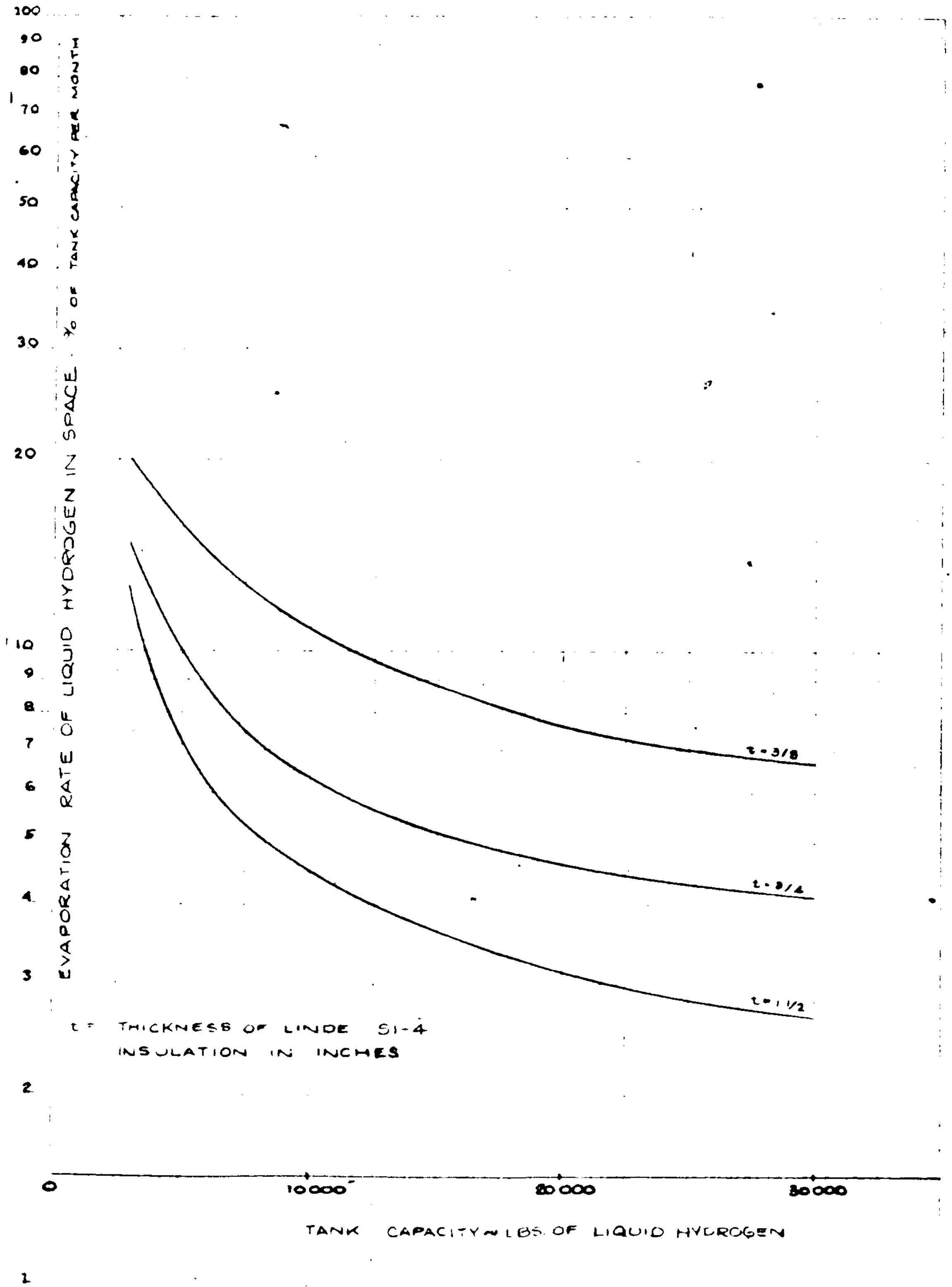

Figure 3-29. Evaporation Rate of Liquid Hydrogen vs Tank Capacity (Ref. 6)

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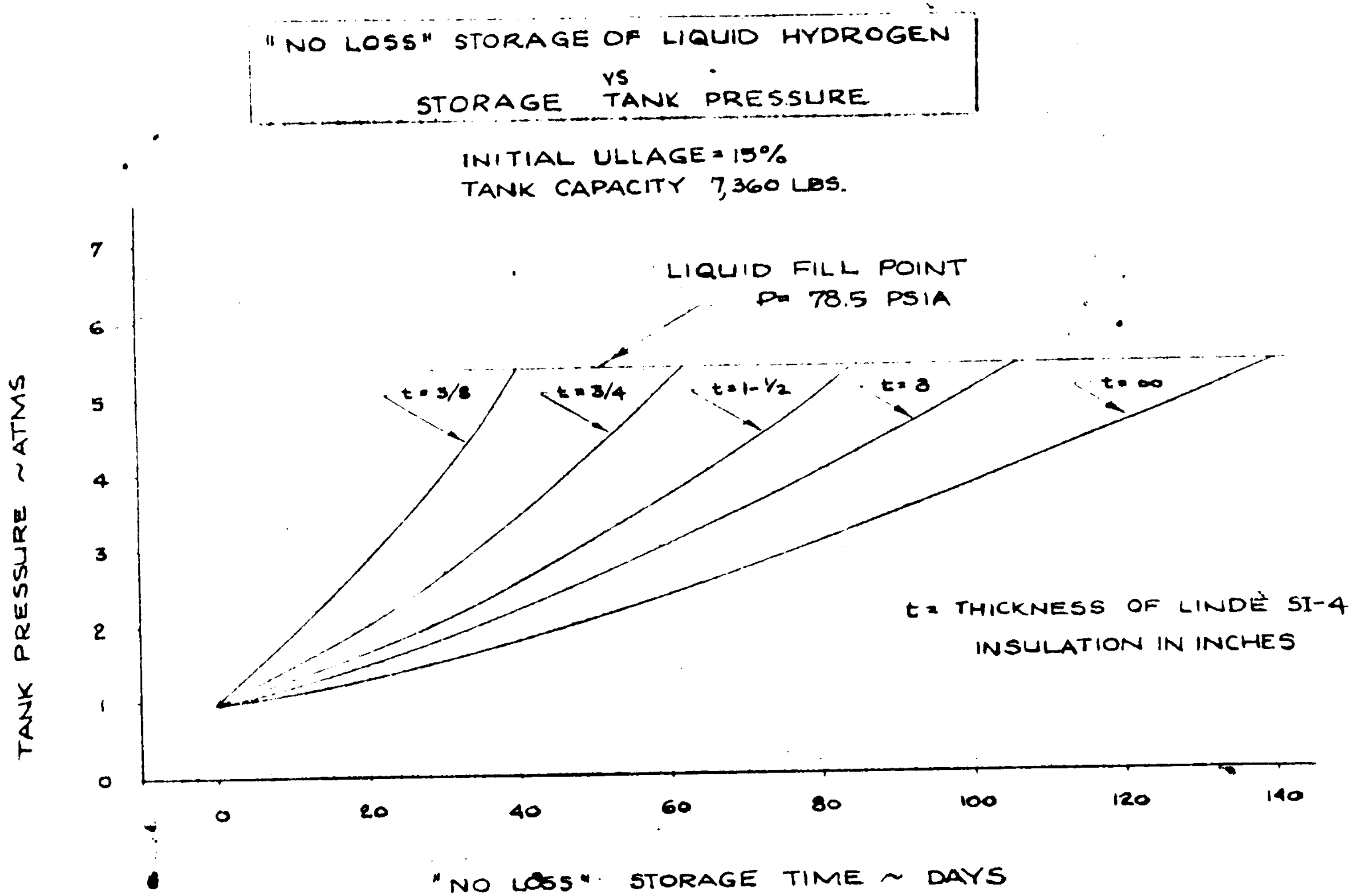

Figure 3-30. "No Loss" Storage of Liquid Hydrogen vs Storage Tank Pressure (Ref. 6)

[redacted]

Considering the various heat flows, both internal and external, and the insulation necessary to control them, a Rocketdyne study (Ref. 3) investigates the effect of storage time on the insulation weight and payload capability of a typical propulsion system based on the previously mentioned propellant temperature limit. Figure 3-31 indicates this effect. This figure shows that for long storage times the insulation required by the cryogenic (high energy) propellants tends to decrease the payload advantage. Eventually, after a very long storage period, Earth-storable propellants provide more payload than the cryogenics. It should be emphasized that these results are for a particular case. Vehicle size, incident radiation intensity, insulation design and internal conduction have a substantial effect on this payload comparison.

In a similar fashion the propellant combinations of LO_2/LH_2 (cryogenic, high energy) and MON/MMH (storable) were studied, parametrically, to determine the storage requirements that are sufficiently strenuous to result in the storable propellant combination providing a greater payload than the corresponding system using cryogenic propellants. This comparison was based upon the pump-fed propulsion systems described below:

Propellant Selection	Thrust to Weight Ratio	Chamber Pressure, psia	Area Ratio	Propellant Tank Pressure, psia	Bulk Density, lb/ft^3	$g_o I_s$ ft/sec
LO_2/LH_2	0.5	500	30	30	20.1	13,800
MON/MMH	0.5	150	25	250	74	10,300

[redacted]

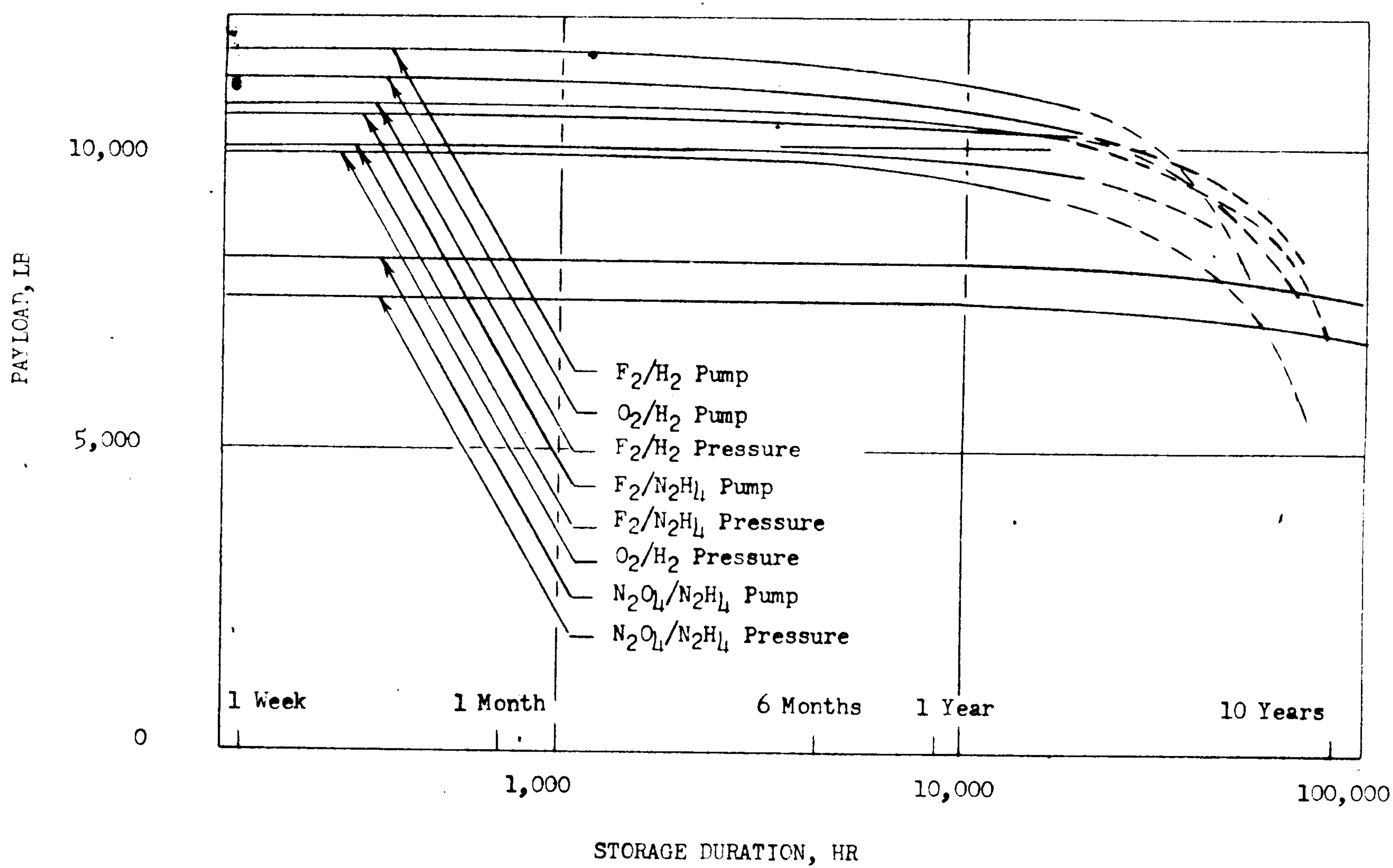

Figure 3-31. Effect of Storage Duration on Payload (Ref. 3)

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The comparison was based on the assumptions that the storable system needs no insulation (freezing neglected) and that the cryogenics are stored with no propellant losses. For a given velocity requirement (ΔV) and propellant weight, the payload-to-gross weight ratio of the MON/MMH stage was determined. For the same ΔV and propellant weight, calculations are then made to determine the amount of insulation that must be added to the LO_2/LH_2 stage to reduce its payload-to-gross weight ratio to the same value. This amount insulation is then translated into terms of storage time. An increase in storage time increases the insulation necessary for propellant maintenance. This increase in the amount of insulation results in the MON/MMH system providing a larger payload-to-gross weight ratio than the LO_2/LH_2 system.

The determination of the insulation requirements as a function of storage time are described in the following paragraphs.

The heat input to the LH_2 was assumed to affect the entire propellant mass. Propellant tank designs were based upon insulation requirements for LH_2 which were then assumed to hold for LO_2. The insulation used for the cryogenic propellant tanks was Linde SI-4, having a thermal conductivity (k) of 2×10^{-5} Btu/hr ft R, and a density of 4.7 lb/ft^3.

Initial tankage conditions for the LH_2 were based upon an initial temperature, of 36.7 R. The LO_2/LH_2 tanks were designed for an internal pressure of 30 psia. This pressure corresponds to the maximum allowable change in vapor pressure resulting from an influx of heat into the liquid hydrogen. The increase in LH_2 temperature, and corresponding decrease in LH_2 density due to heat influx determined the cryogenics ullage requirements of 12 percent of the propellant volume. The following table gives the initial and final LH_2 characteristics leading to the design criteria for the propellant tanks.

	Initial Conditions	Final Conditions
Tank pressure, psia	14.7	30
Temperature, deg R	36.7	41.2
Vapor pressure, psia	15.8	29.5
Density, lb/cu ft	4.42	4.26
Enthalpy, Btu/lb R	116	131

The enthalpy change of 15 Btu/lb was then equated to the heat input per unit of time entering the propellant tanks.

$$\dot{Q} = \frac{\Delta h \, W_p}{\theta} \tag{1}$$

where

$\dot{Q}$ = Heat rate, Btu/hr

Δh = Enthalpy change, Btu/lb

W_p = Propellant weight, lb

θ = Heat transfer time, hours

Equation 1, which is the total heat influx into the propellant tank is a result of radiation on the outer skin surface and heat conduction between the LH_2 and LO_2 propellant tanks. This can be expressed by; the simplified expression

$$\dot{Q} = \frac{\Delta h \, W_p}{\theta} = \frac{k \, A \, \Delta t}{t} + \dot{Q}_r \tag{2}$$

where

k = Insulation conductivity, 2×10^{-5} Btu/hr sq ft R

A = Surface area, sq ft

To estimate the temperature change (ΔT) between the outer skin surface and inner tank surface a mean skin temperature to 300 R was assumed to be achievable through proper attitude control and surface conditions. The heat conduction rates were assumed to be 0, 10, and 100 Btu/hr.

From this equation the maximum allowable storage time (θ) is found for a given insulation thickness and heat conduction rate between the propellant tanks. As described previously the effects of insulation thickness on system performance can be determined.

The results of these calculations are presented in Fig. 3-32, 3-33, and 3-34 for heat conduction rates of 0, 10, and 100 Btu/hr respectively. These plots represent the combinations of storage time, propellant weight, and velocity requirement (ΔV) that result in the MON/MMH combination providing the same payload-to-gross weight ratio as the LO_2/LH_2 system. It is apparent that if the propellant weight is decreased or the storage time increased above the values plotted the MON/MMH system is superior to the LO_2/LH_2 system.

The regions where each propellant combination can be used to advantage are indicated. For example, for a propulsion system with about 100,000 lb of propellant that must be stored for a year assuming a heat conduction rate of 100 Btu/hr the MON/MMH combination will be most attractive. If the storage time is somewhat less, say one month, the LO_2/LH_2 system will be more attractive.

The extreme significance of the assumed heat conduction rate in determining these regions is apparent. This conduction is a function of the detailed system design. As such it is difficult to estimate the value to be used in a general study.

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Heat conduction between propellant tanks = 0 BTU/hr.
No storage losses.
Pump-fed systems.
LINDE SI-4 insulation.

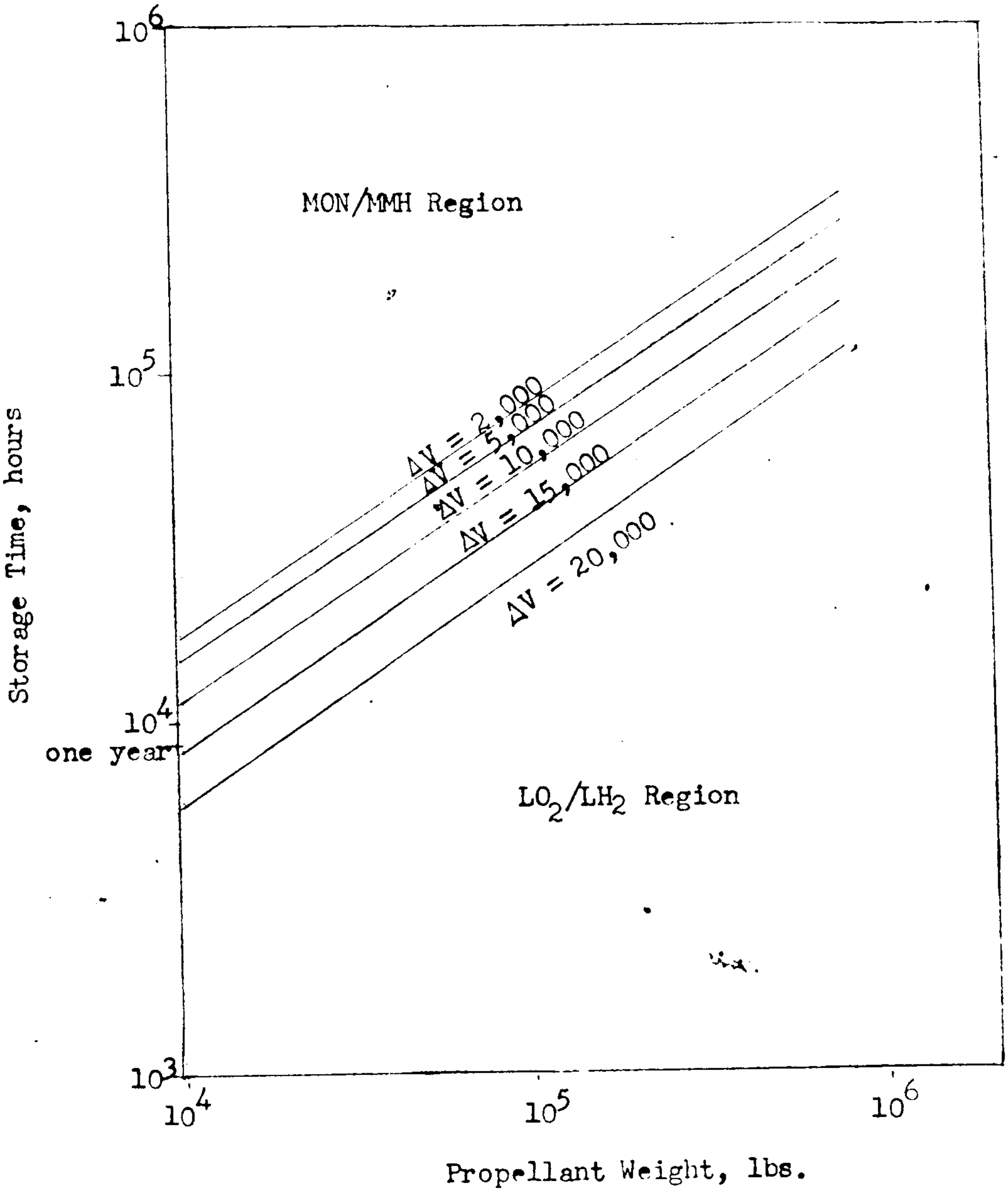

Figure 3-32. Effect of Space Storage Time on Propellant Selection

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Heat conduction between propellant tanks = 10 BTU/hr.
No storage losses.
Pump-fed systems.
Linde SI-4 insulation.

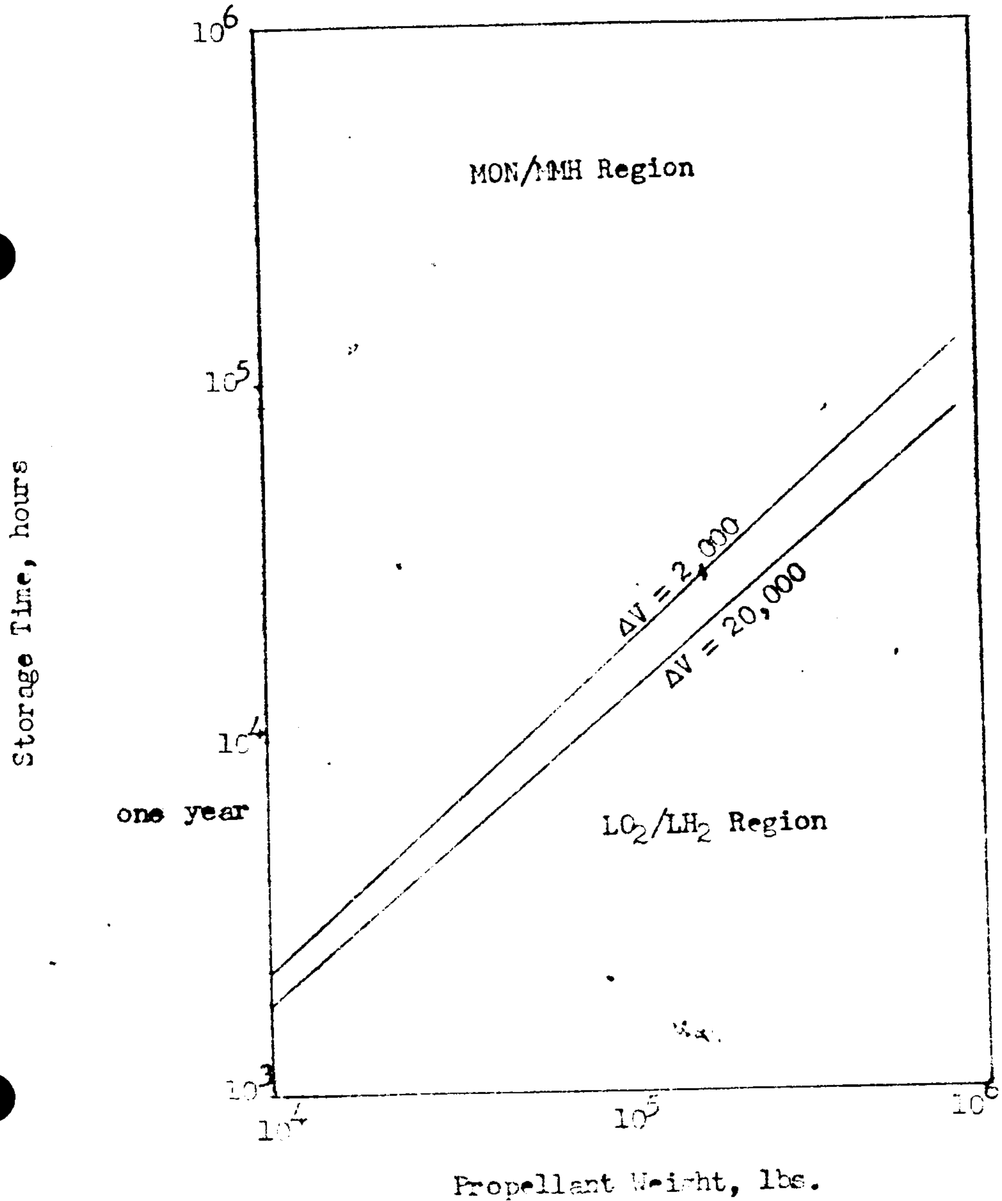

Figure 3-33. **Effect of Space Storage Time on Propellant Selection**

[redacted]

Heat conduction between propellant tanks = 100 BTU/hr.
No. storage losses.
Pump-fed systems.
LINDE SI-4 insulation.

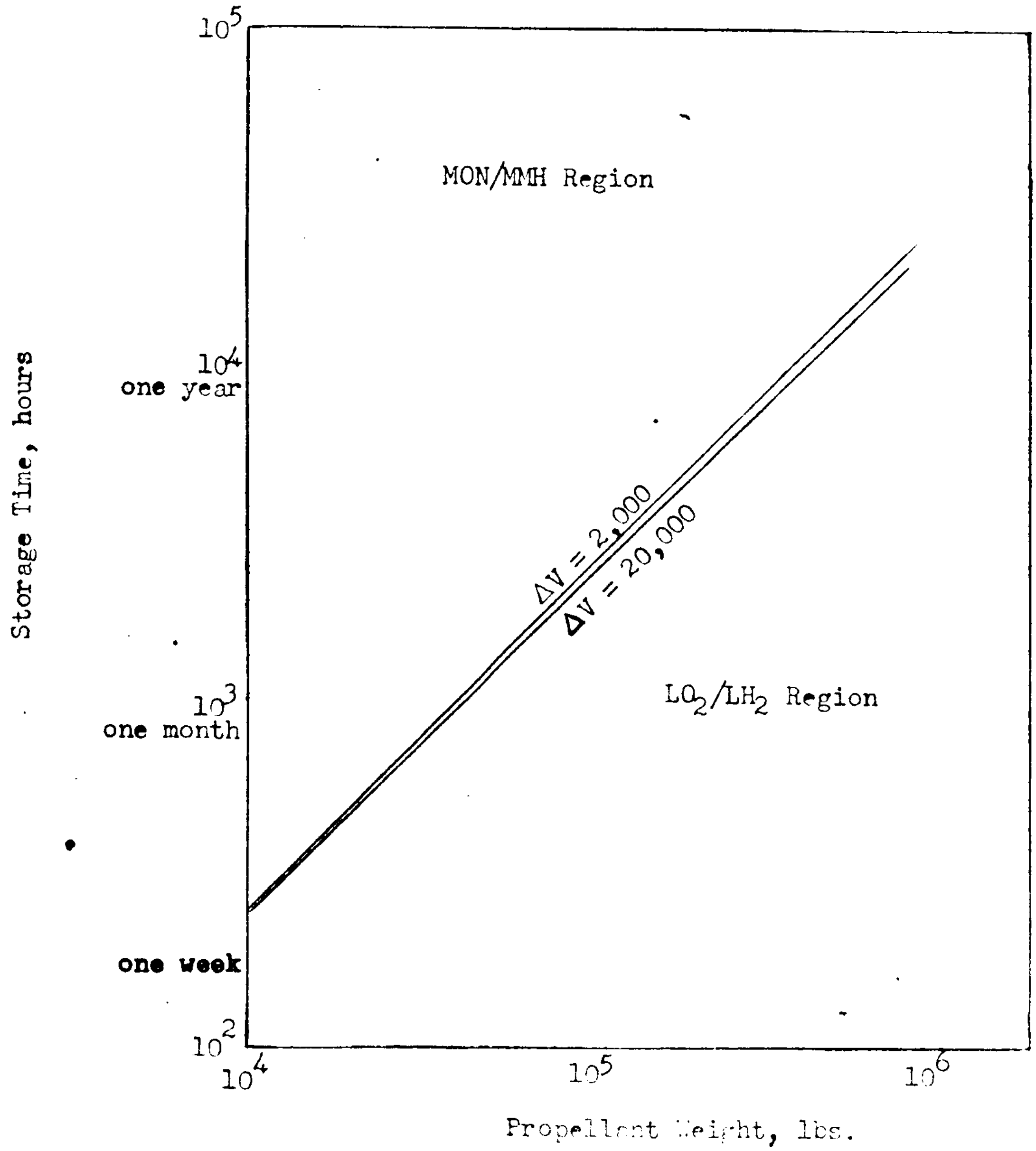

Figure 3-34. Effect of Space Storage Time on Propellant Selection

The regions in which the two propellants are desirable would also be affected if some of the hydrogen were allowed to boil off. It is possible that longer storage times for the LO_2/LH_2 combination could be achieved through some combination of "no loss" and "boiloff" storage.

Meteoroid Environment

The effect of the meteoroid environment on the propulsion system is essentially the same as that discussed in the payload section. Puncture of propellant tanks, thrust chamber walls, or any of the engine system components may render a propulsion system inoperable. Figure 3-35 from Ref. 7 indicates a distinct possibility of meteoroid penetration. It is therefore apparent that some form of protective system will be necessary.

At the present the most promising protection system is the "Whipple meteor bumper." The bumper consists of a thin shield, spaced a small distance from the object to be protected. A particle hitting this outer shield is shattered and, although the bumper is penetrated, the penetration of the protected wall is considerably reduced. Reference 8 indicates that for lead materials, use of a 0.075-in. shield, 1/2 in. from the target, reduced target penetration to 20 percent of the unshielded penetration.

Figure 3-36 from Ref. 9 shows the ratio of wall thickness to particle diameter as a function of the ballistic limit. The ballistic limit is defined as the particle velocity resulting in second plate penetration such that a one atmosphere pressure difference cannot be maintained across the second plate. Using this figure (extrapolated) and a typical meteoroid velocity of about 100,000 fps, a wall thickness to particle diameter ratio of about 3.2 is estimated. If, as in the payload design section, a meteor

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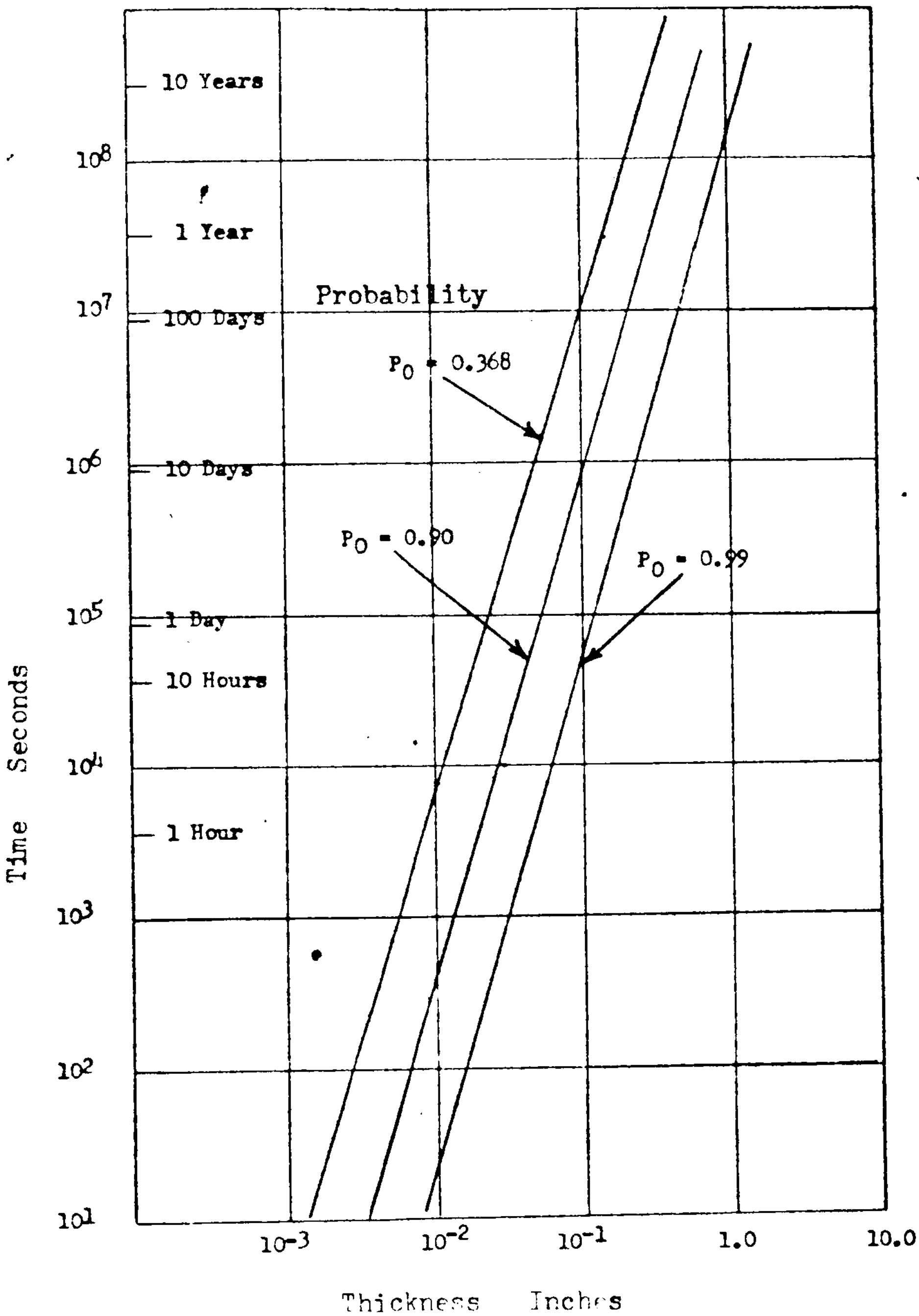

Figure 3-35. Probability of No Meteroid Penetrations in 100 Square Feet of Surface Area as a Function of Time in Space and Wall-Thickness, Calculated From Estimates of Whipple and Bjork (Ref. 7)

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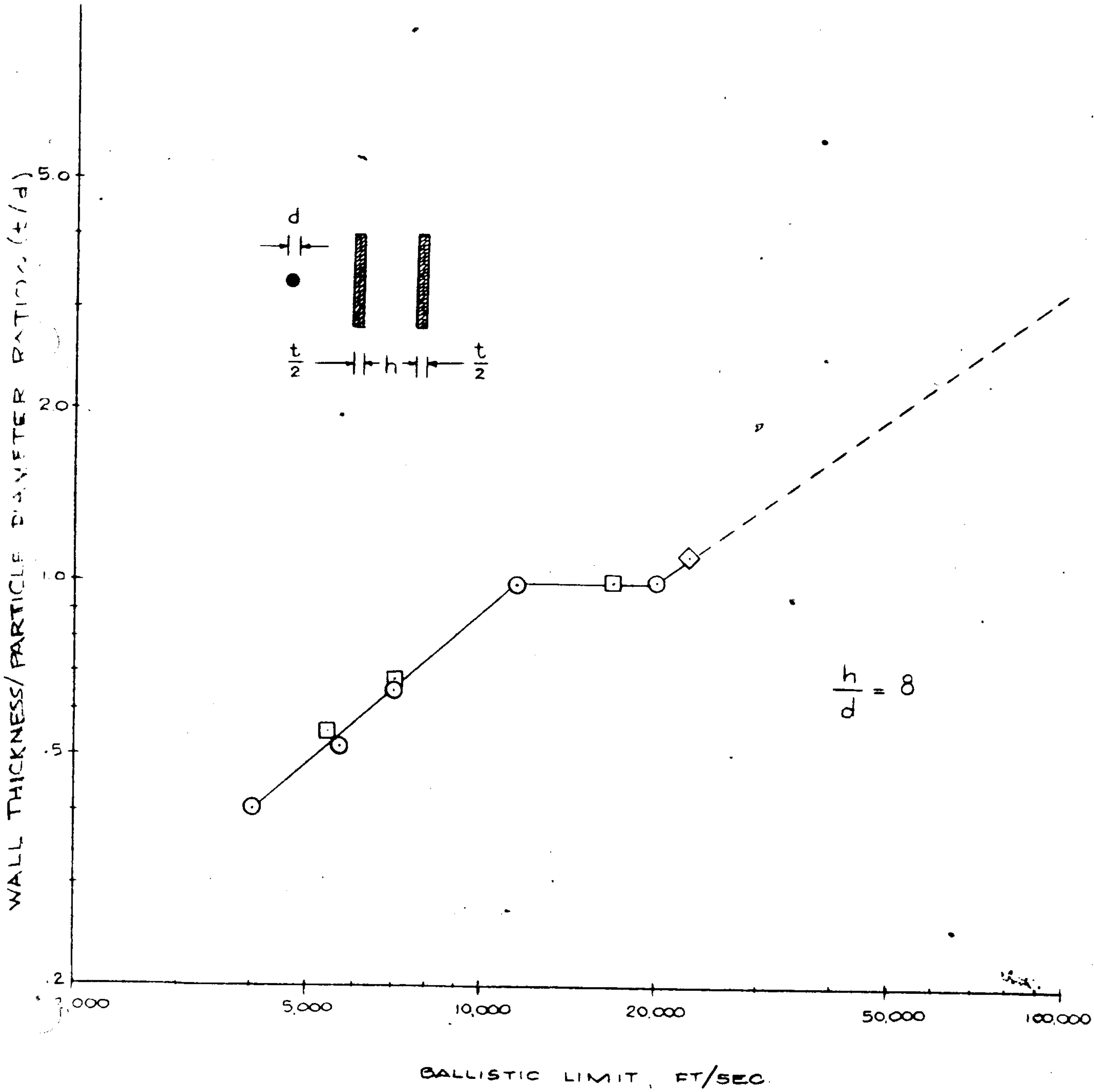

Figure 3-36. Effect of Thickness on Ballistic Limit (Ref. 9)

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of magnitude 6 (diameter of 0.0244 in.) is selected for design purposes, the required shield thickness (t/2) is 0.039 in.

Reference 9 also indicates that the use of a low density filler between the two walls will substantially increase the protective ability of the outer wall. For the propulsion systems considered in this study the Linde SI-4 insulation mentioned previously was assumed to suffice as a filler material. Placing an aluminum shield of 0.032 in. over this insulation was assumed to provide sufficient protection. Thus, for preliminary estimates the meteor bumper system would weigh about 0.45 lb/sq ft of surface area. The filler (insulation) weight is not included, and would be determined from the insulation requirements.

Vacuum Effects

The effects of "hard" vacuum, such as that encountered in space, on propulsion system operation are many. The low pressures encountered (on the order of 10^{-12} in. Hg can cause pernicious effects which are not normally encountered even in the laboratory. The absence of damping of vibration, explosive decompression, vaporization of materials, and various surface effects offer many potential problems which must be taken into account in propulsion system design.

Under the exposure to high vacuum some materials vaporize to an extent that their usefulness is significantly affected. Some metals such as magnesium, and various organic materials such as neoprene, epon potting compounds, and MIL-D-109240 grease vaporize to a considerable extent over prolonged high-vacuum exposure, particularly at high temperatures. Information on many of these compounds is available in the literature, and can be taken into account in propulsion system designs.

Table 3-5 shows an estimated order of merit for behavior of plastics in a vacuum. Temperatures at which the estimated weight loss is 10 percent per year are given where available, but are subject to considerable uncertainty, particularly in the case of the rubbers. Maximum recommended service temperatures in air are also given for comparison. In most cases, these are based on mechanical considerations, and will presumably apply in a vacuum. High polymers are lost in vacuum, not be evaporation, but by breaking down of the chains into smaller fragments. This process is often accelerated by small amounts of impurities and addition agents, including polymerization catalysts.

Plasticizers and mold lubricants are also highly detrimental to stability in vacuum, which therefore may be strongly affected by the particular formulation and curing procedure used.

Table 3-6 gives the probable maximum temperatures at which metals and semiconductors can be used in space vacuum environment, assuming three limiting evaporation rates. These temperatures were calculated on the basis of Langmuir's equation

$$P = 17.14\, W \sqrt{T/M}$$

where

P = Vapor pressure, mm hg

W = Rate of evaporation, gm/sq cm-sec

T = Temperature, deg K

M = Molecular weight

TABLE 3.5

PLASTICS IN SPACE ENVIRONMENT

	Vacuum Temperature at Which Loss is 10 percent/year, deg F	Air Reduction Recommended Service Temperature, deg F
Polysulfide	...	250
Cellulose nitrate	140	130
Polyester	...	250
Epoxy-polyamide	...	300
Acrylate (commercial coating)	...	...
Cellulose acetate butyrate (plasticized)	...	225
Cellulose acetate (plasticized)	...	300
Vinyl chloride	...	180
Polyurethane (cured at 50 to 100-percent humidity)	...	240
Vinyl butyl	...	115
Silicone alkyd	...	...
Linseed oil (alkali refined)	...	...
Chloroprene (neoprene)	...	240
Alkyd phenolic	...	300
Epoxy-amine	...	300
Methyl methacrylate (benzoyl peroxide catalyst)	220	165
Polyurethane (cured at 0 to 20-percent humidity)	...	...
Cellulose (with NaCl)	270	...

TABLE 3-5

Continued

	Vacuum, Temperature at Which Loss is 10 percent/year, deg F	Air Reduction Recommended Service Temperature, deg F
Cellulose acetate butyrate (no plasticizer)	...	225
Styrene	280	160
Nylon	300	250
Phenolic	...	400
Methyl methacrylate (diphenyl cyanomethyl catalyst)	320	...
Polycarbonate	...	...
Methyl acrylate (benzoyl peroxide catalyst, no other additions)	320	....
Rubber natural	360	180
Isoprene	360	...
Cellulose (pure unoxidized, no plasticizer)	360	375
Methylstyrene	370	...
Cellulose acetate (no plasticizer)	370	...
Ethylene terephthalate (mylar, dacron)	390	...
Isobutylene	400	...
Butadiene	...	...
Butadiene-styrene (GR-S = SBR)	...	180
Polypropylene	...	...
Methyl methacrylate (no catalyst)	410	...

TABLE 3.6

EVAPORATION OF METALS AND SEMICONDUCTORS IN HIGH VACUUM

Element	Temperature (deg F) at Which Evaporation is			Melting Point, F
	10^{-5} cm/yr	10^{-3} cm/yr	10^{-1} cm/yr	
Cadmium	100	[illegible]	250	610
Selenium	120	[illegible]	240	430
Zinc	160	260	350	790
Tellurium	260	350	430	840
Magnesium	260	350	460	1200
Lithium	300	[illegible]	530	370
Antimony	[illegible]	[illegible]	570	1170
Bismuth	[illegible]	600	750	520
Lead	510	630	800	620
Indium	760	940	1130	310
Manganese	845	1010	1200	2300
Silver	890	1090	1300	1760
Tin	1020	1220	1480	450
Aluminum	1020	1260	1490	1220
Beryllium	1140	1300	1540	2340
Copper	[illegible]	1400	1650	1980
Gold	1220	[illegible]	1750	1950
Germanium	1220	[illegible]	1750	1760
Chromium	1380	1600	1840	3440
Iron	1420	[illegible]	1920	2800
Silicon	[illegible]	[illegible]	1970	2580
Nickel	1480	[illegible]	2000	2650
Palladium	1490	[illegible]	2020	2840

TABLE 3-6

Continued

Element	Temperature (deg F) at Which Evaporation is			Melting Point, F
	10^{-5} cm/yr	10^{-3} cm/yr	10^{-1} cm/yr	
Cobalt	1500	1760	2020	2720
Titanium	1690	1960	2280	3140
Vanadium	1870	2150	2460	3100
Ruthenium	2080	2420	2800	3580
Platinum	2120	2440	2840	3240
Boron	2240	2580	2980	3720
Zirconium	2340	2740	3150	3360
Iridium	2380	2740	3150	4450
Molybdenum	2520	2960	3450	4700
Carbon	2780	3050	3400	6600
Tantalum	3250	3700	4200	5700
Rhenium	3300	3700	4200	5700
Tungsten	3400	3900	4500	6100

Mechanical motion in a vacuum necessitates the recognition of two major effects: (1) greases and other conventional lubricants may change their characteristics or disappear altogether due to volitilization, and (2) dry sliding friction in some combinations is much higher than in air due to the removal of normal surface film of absorbed air. The evaporization of this surface layer of air has an interesting effect in that the situation arises where a "perfectly clean" surface must be considered. Two such clean surfaces in contact have a very strong tendency to cold weld together at the very small areas or points that actually touch each other.

Increases in dry coefficient of friction of 300 to 500 percent in vacuum have been experimentally demonstrated. Lubrication is complicated by the normal evaporation of lubricants, many of which have high vapor pressures, and by the dependence of many surfaces upon layers of absorbed gas for additional lubricating effects. Desorption of gas in a high vacuum may cause such surfaces to seize, gall, and weld. Low vapor pressure lubricants, such as molybdenum disulfide, are satisfactory if used in amounts large enough to prevent complete evaporation of the lubricant during the service life of the part. Vacuums in the order of 10^{-6} in. of mercury have been observed to reduce the service life of such lubricants by a factor of 10.

High vacuums encountered in outer space (10^{-12} to 10^{-17} in. of mercury) may affect certain electronic components whose electrical properties depend upon their surface crystal structure. Mica and asbestos, for example, depend upon loosely bonded water of hydration for their dielectric properties. These weakly bonded surface molecules may escape in high vacuum. Where their use is mandatory, pressurized containers may be required.

The fatigue life and creep phenomona of metals at reduced pressure also show interesting differences from their performance at ordinary pressures. The fatigue life of some metals seems to increase at low pressures. Creep ruptured curves for low-term tests in air and vacuum for nickel seem to indicate a shorter time to rupture for the vacuum tests.

The spectral emissivity of the surface of the propulsion system is a critical factor in determining thermal balance as shown in the previous section. This emissivity is strongly dependent on the condition of the surface of the material. As the high vacuum of space causes the surface to change due to volitization of the surface layer of air and vaporization of material, the emissivity is likely to change also.

On the basis of the information available, it appears that problems caused by high vacuum can be avoided in the system by careful selection of materials in critical applications. The uncertainty of much of the data, however, makes it highly desirable that these applications be confirmed by environmental testing.

Particulate Radiation Effects

The interaction of energetic radiation with matter is complex. The most serious effects resulting from this radiation are biological. These effects are discussed in detail under the payload design criteria considerations. Next to human beings, semi-conductor materials used in transistors are most susceptible to damage from radiation. Specific effects which might be encountered in the propulsion system are discussed below.

Table 3-7 lists the relative radiation stability of various plastics. Curves of variation in physical properties as a function of absorbed radiation dose are also available for most substances of interest.

The reaction of Teflon to nuclear radiation is well documented. It is one of the more sentitive plastics, tending to become brittle, to fragment or powder, or to release fluorine gas when irradiated. It becomes seriously degraded by relatively low radiation doses (5×10^4 rads) and may therefore be considered to be marginal in suitability.

Mylar, polyethylene-terrepthalate, a polyester, begins to change its properties at 10^5 to 10^6 rads. The addition of mineral fillers is said to increase the radiation stability of polyesters 100-fold. This increase in resistance to radiation by filling is generally true for most plastics.

Table 3-8 shows the relative stability ratings for elastomers. Butyl rubber, has the least radiation stability of any of the common synthetic rubbers. It may be noted from Table 3-8, however, that its damage threshold is well above the anticipated dose. Butyl is said to retain fair properties even after a dose of 1×10^6 rads.

Silicones of all types are said to accrue 25-percent damage at a dose of 5×10^4 rads. With such severe damage at this radiation level, some deterioration can be expected from the anticipated dose of 6×10^2 rads. Thus, silicones are considered of marginal suitability.

Buna N, an acrylonitrille-butadiene copolymer, is stated to retain fair properties after doses of 1×10^7 rads. The threshold of damage is quoted as 2×10^6 rads. The anticipated dose, 6×10^2 rads is well below this threshold, so no problem is apparent.

TABLE 3-7

RELATIVE RADIATION STABILITY OF PLASTICS

Class I	-	Retain fair properties of doses of 10^8 rads	
		Phenolics (filled)	Epoxies
		Polystyrene	Silicones (filled) e.g., DC-122
		Polyesters (filled) e.g. Mylar	
Class II	-	Retain fair properties after doses of 10^9 rads	
		Polyethylene	Polyesters (fibers)
		Polyvinyl chloride	Phenolics (unfilled)
Class III	-	Retain fair properties after doses of 10^6 rads	
		Polyamides	Acrylics
		Silicones (unfilled)	Polyformaldehyde
		Polyesters (unfilled)	
Class IV	-	Retain fair properties after doses of 10^4 rads	
		Teflon	

TABLE 3.8

RELATIVE RADIATION STABILITY OF ELASTOMERS

Class I - Retain fair properties after doses of 10^8 rads

- Natural rubber
- Adduct rubbers
- Polyurethanes
- Styrene butadiene copolymers (Buna S)
- Vinyl pyridine elastomers

Class II - Retain fair properties after doses of 10^7 rads

- Acrylonitrile butadiene copolymers (Buna N)
- Polybutadiene
- Neoprene
- Hypalon
- Polyacrylates
- Silicones
- Fluorocarbon elastomers
- Polypropylene polyethylene elastomers

Class III - Retain fair properties after doses of 10^6 rads

- Thiokol
- Butyl

Many of the effects resulting from particulate radiation concern insulation and thermal balance of the propulsion system. Exposure to this radiation for long periods may influence the behavior of the outer most layer of external surface of the vehicle. This roughening on a microscopic scale may affect the emissivity of a metallic surface and thus the thermal balance of a propulsion system. The thermal insulation of the propulsion system may be affected through out-gassing of the surfaces, thus raising the gas pressure in the insulation and causing its effectiveness to deteriorate.

The absorption of radiant energy by the propellants in a chemical propulsion system may lead to a variety of chemical changes. These changes vary from system to system, and depend upon the total dose radiation. Estimates of these effects in liquid oxygen and liquid hydrogen indicate that they are negligible.

Zero Gravity Effects

In the unpowered portions of space flight when the vehicle is moving solely under the action of gravity, the propulsion systems will be exposed to the effects of weightlessness or zero gravity. The term zero gravity is used since the space vehicle and its contents are both experiencing the same gravitational acceleration, and therefore the acceleration of the contained matter relative to the vehicle is zero.

The primary effects of zero gravity are due to the absence of any body forces. Each body retains its own mass, but does not exert any weight force on its environment. For example, in a liquid with a free surface such as found in the propellant tanks of a space vehicle, there would be no hydrostatic force. Therefore, buoyancy and natural convection would not exist.

Processes which are sensitive to a gravity gradient are outlined below:

1. Hydromechanics

   a. Hydrostatics: behavior of liquids stored in containers with ullage

   b. Hydrodynamics: flow of liquids and liquid-gas mixtures through pipes, nozzles, pumps, turbines, and separators

   Lubrication: rotating and sliding bearings with fluid lubricant

2. Fluid Heat Transfer

   a. Stationary systems: single phase (free convection); two-phase pool boiling and condensing

   b. Flow systems: single phase forced convection in heat exchangers; two-phase forced circulation evaporation in boilers and condensation in condensers

The studies directed toward discerning these effects are discussed at length in Ref. 10 through 12. This discussion will be briefly summarized along with some design methods for eliminating the problems arising from these effects.

Hydrostatics. The hydrostatic behavior of liquids at or near zero gravity has been subjected to numerous analyses which take into account deformations due to adhesive and cohesive forces. Many analyses have been concerned with determining the shapes that fluids assume if left to themselves; for example, the fluids in a propellant tank which are subjected to zero gravity.

A theoretical analysis (Ref. 11) has shown, using the principle of minimum surface energy, that the stable configurations of a bubble will be a surface of revolution if allowed by the container walls. Vapor bubbles will coalesce to form a larger bubble, which will eventually join the central bubble by contact. It concludes that zero gravity hydrostatics are dominated by surface tension, and the stabilizing effect of small rotational disturbances will cause bubbles to coalesce along the center of rotation.

The analyses have in general been verified by experiment. These experiments have shown that in a propellant tank with ullage the wetting liquids (water, oil, etc.) will crawl around the tank walls, leaving a gas pocket in the center. Non-wetting fluids will coalesce in the center of the tank leaving a complete gas blanket between the propellant and the walls.

Since cryogenic liquids are more wetting than water, the effects described above will be seen to a greater degree. The behavior of a cryogenic liquid in a steel tank will be dominated by surface tension when exposed to zero gravity. Its equilibrium configuration will be a surface of minimum area that is uniquely determined by the shape of the container, the contact angle of the fluid interface with the container walls, and the ratio of the volumes of the fluids in the tank. Sketches of some configurations are shown in Fig. 3-37.

A major problem created by these effects is that of separating the gases and liquids in the tanks during propulsion system start. A number of methods of accomplishing this are available. Two straightforward methods are:

1. Accelerating the tank with an auxiliary rocket; the liquid

[redacted]

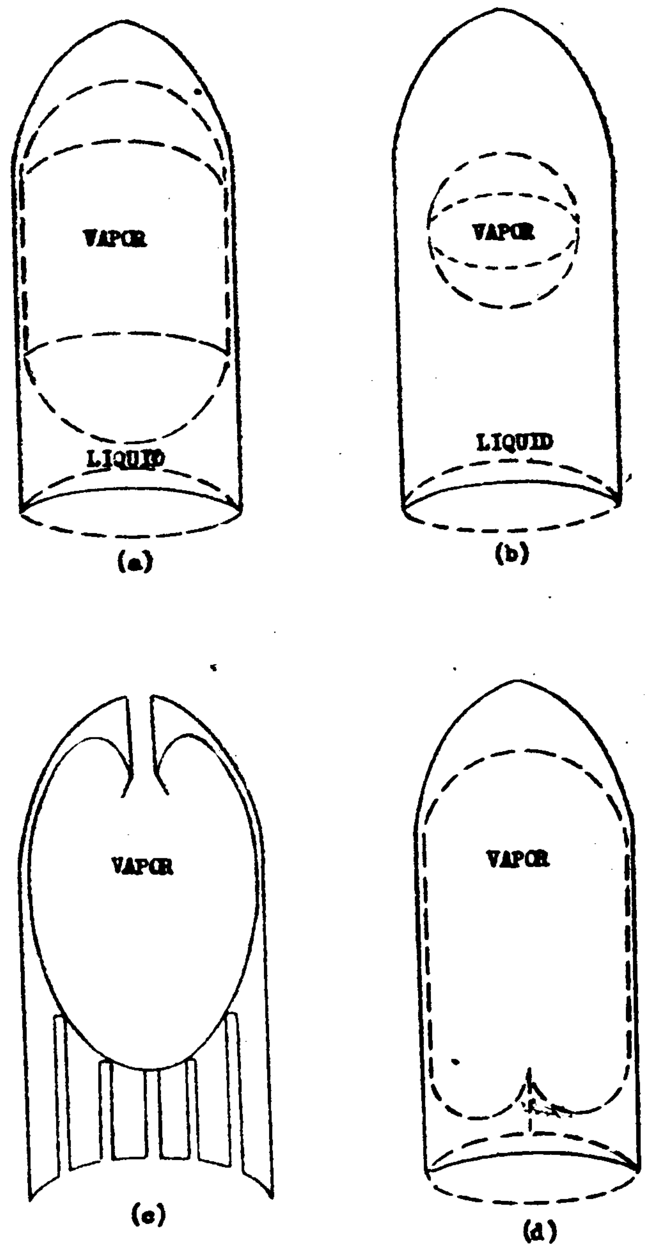


Figure 3-37. Hydrostatic Behavior of a Contained Liquid Under Zero Gravity Conditions (Ref. 11 and 12)

[redacted]

gas interface will be perpendicular to the acceleration

2. Permanent separation of the liquids and gas as with a diaphgram

A somewhat subtler method is suggested in Ref. 12 and uses the wetting properties of the cryogenics. Poles are planted on the floor of the tank (Fig.3-37 c). Because of the wetting properties of the cryogenic liquid these poles will not readily penetrate the liquid-gas interface. The gas bubble will be kept in the upper part of the tank and the liquid in the bottom. These wetting properties can also be used in designing an entrance to the propellant feed line that will tend to admit only liquids.

Hydrodynamics

An obvious hydrodynamics problem under zero gravity is the separation of liquid and gas phases in two-phase flow. This problem would occur in the venting of cryogenic tanks. An interesting device is described in Ref. 12 which states:

> "Venting devices, utilizing pressure difference, may be used to solve the venting problem. A pipe leading from the inside of the tank to the outside may be equipped with a movable lip, which has the shape of a truncated cone (Fig. 3-37(c)) so that the surface of the minimum area cannot be tangent to the lip. The lip rotates because of the pressure difference. When it rotates a centrifugal force is created which produces a small gravitational field to throw the liquid against the tank wall and to draw the vapor bubble towards the axis. This device will eliminate at least partly the escape of the liquid. It may adequately solve the venting problem."

[redacted]

Lubrication

The effect of zero gravity on linear motion bearings and thrust bearings seems to present no additional operating problems. On the other hand, the effect of zero gravity on radial hydrodynamically operated bearings is very definite and results in the instability associated with unloaded bearings.

This phenomenon can be avoided or controlled by (1) adding external or artificial loading to set the speed at which whirl commences sufficiently higher than the operating speed, (2) increasing the bearing operating clearance to increase the journal eccentricity, and hence the speed at which a significant unstable zone is created, and (3) designing a stable geometry using grooves and/or multiple pads.

Heat Transfer

During periods of zero gravity heat transfer will be occurring between the propellants and the propellant tank walls. The type of heat transfer and the heat transfer coefficients will be important in the determination of propellant heating. Experiments have demonstrated that free convection is absent during zero gravity conditions. Film boiling experiments under zero gravity conditions have indicated that once the transient gravitational effects have ceased, the zero gravity and unit gravity heat fluxes are very similar.

[redacted]

ESTIMATES OF CUT-OFF IMPULSE DEVIATION FOR FUTURE SPACE ENGINES

In the course of rocket engine operation, transient thrust conditions occur at engine start and cut-off. A typical thrust history illustrating this is shown in Fig. 3-38. These transient conditions can lead to inaccuracies in position and velocity at the end of the powered flight phase, resulting in deviations from the desired mission profile.

In general, the effects of the startup transient can be modified during the steady-state engine operation. The guidance system will be aware of the impulse contribution of the start transient and will regulate the cut-off time accordingly. An allowance can also be made for a predicted nominal impulse contribution during the cut-off transient (cut-off impulse). However, variations in engine parameters during cut-off cause deviations in cut-off impulse from this nominal value. It is the purpose of this study to estimate the variations in cut-off impulse which can be expected in future engines. These estimates can then be used in studies to determine their effect on the flight path of future space missions. The propellant combinations of liquid oxygen/liquid hydrogen and mixed oxides of nitrogen/monomethyl hydrazine (MON/MMH) are considered.

Analysis

Cut-Off Impulse. The thrust termination transient and resulting cutoff impulse which occur at engine cutoff can be attributed to several factors. First, valve closure is not instantaneous. Therefore, propellant continues to flow and burn at a decreasing rate during the valve

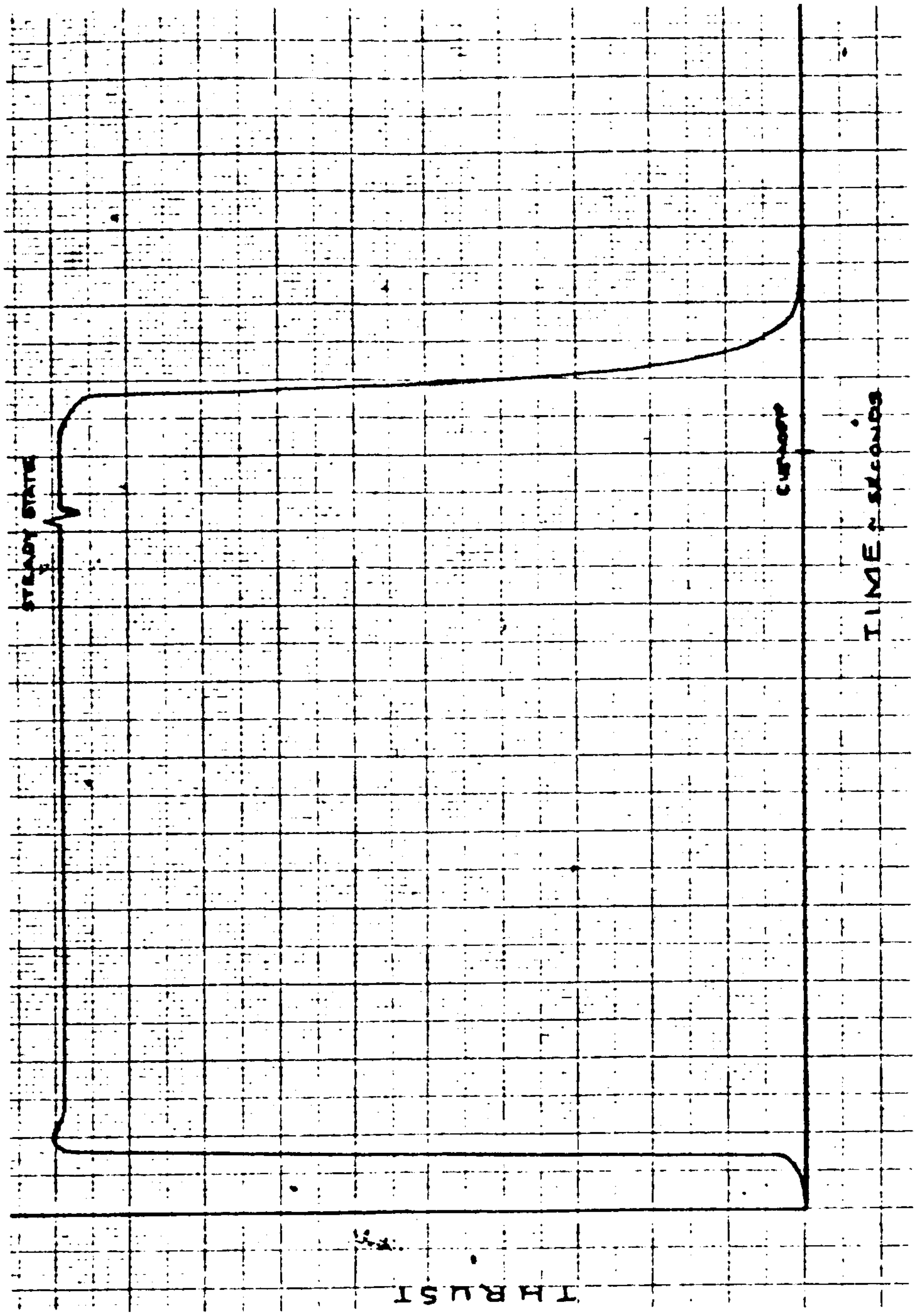


Figure 3-38. Typical Thrust Buildup and Cutoff Transients

closing. Second, after valve closure, a quantity of propellant remains in the cooling jacket, manifolds, etc. This continues to burn, providing additional impulse. Third, upon completion of this burning, the gases remaining in the combustion chamber are exhausted until the chamber pressure is equal to ambient. The contributions of these factors to the total cutoff impulse depends upon the propulsion system design.

If the cutoff impulse could be duplicated on a run-to-run basis, the space vehicle and its guidance could simply be designed to consider this contribution in any given mission. However, variation in engine parameters cause the cutoff impulse to deviate from the nominal value. If the effects of this deviation on the mission are significant, they may be eliminated by either reducing the variation to an acceptable level through a more extensive design and test effort or by providing a compensating effect in a later propulsion phase.

Deviations in cutoff impulse considered in this study can be attributed to variations in three engine parameters: (1) thrust, (2) actuator time, and (3) main propellant valve closing time. The manner in which these factors individually affect the thrust termination transient and, therefore, cutoff impulse is illustrated in Fig.3-39(a). In practice, these effects would be summed, resulting in the total cutoff impulse deviation.

In acceptance testing of a rocket engine, the calibrated thrust is generally required to be within ± 3 percent of some nominal value. In ordinary test procedure, the thrust of a single specific engine, considering run-to-run variations, will be within ± 1 percent. This is illustrated on the following page.

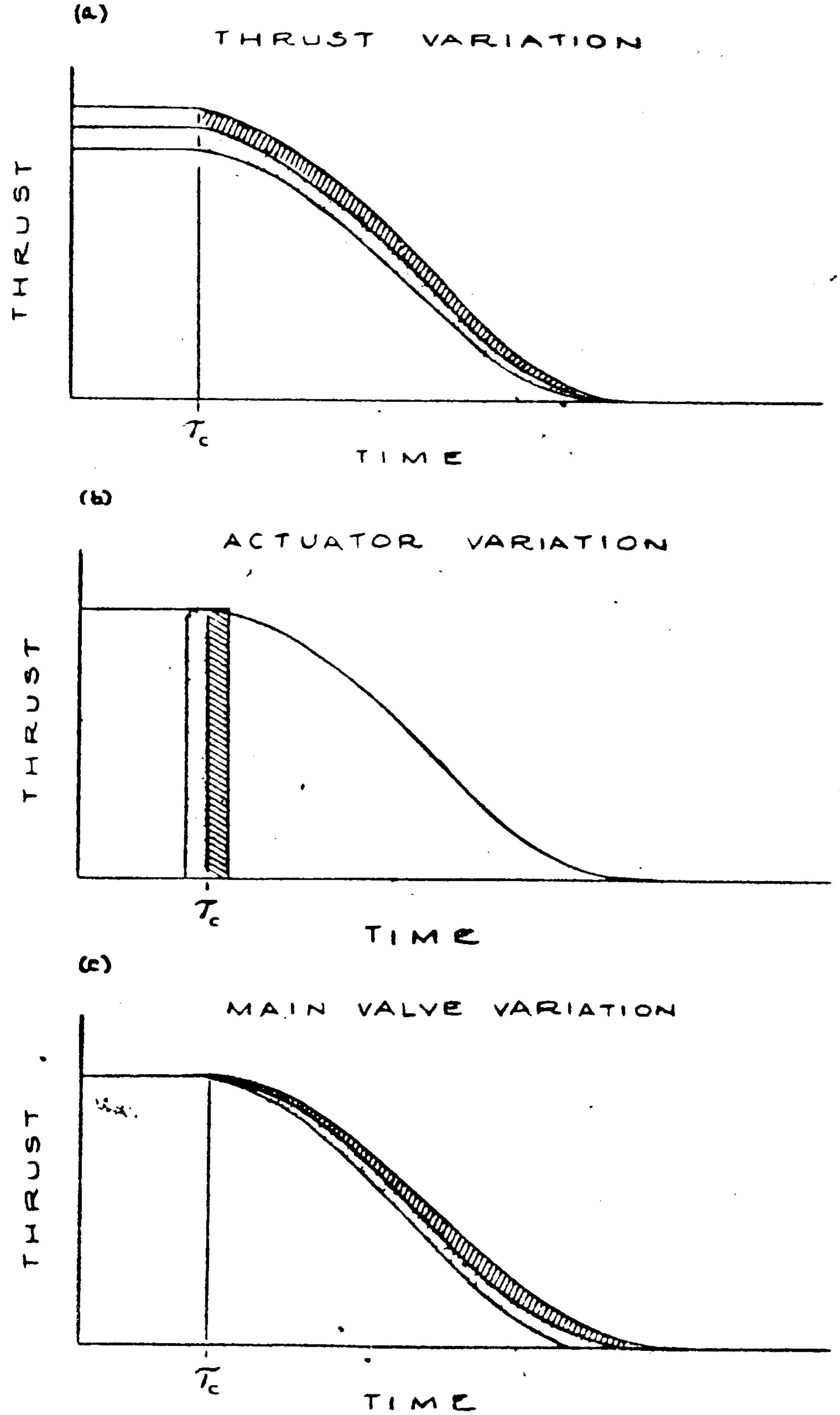
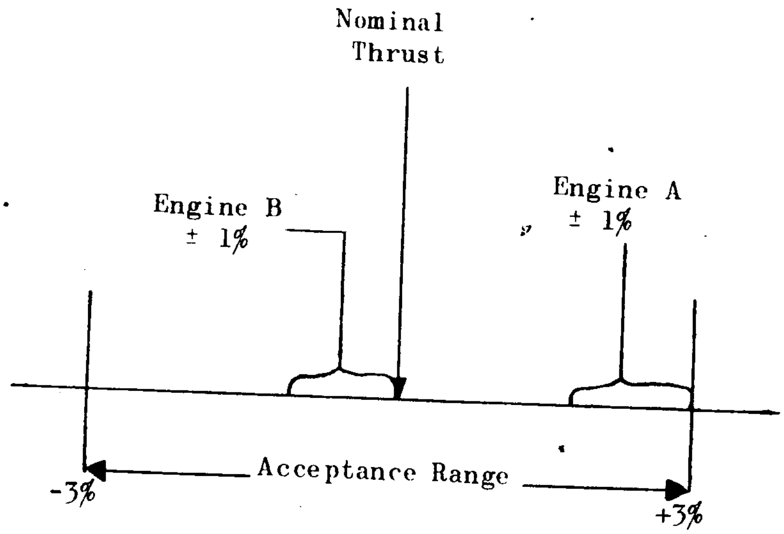


Figure 3-39. Variations in Cutoff Impulse

[redacted]

Thrust Calibration



This 1 percent run-to-run variation could be reduced through an extensive test and calibration procedure for each engine. Factors other than calibration accuracy contribute to variation in thrust level. Change in propellant density, feed system inlet pressure, mixture ratio, etc., lead to the deviation of thrust level from the nominal value.

The contribution of thrust level variation to cutoff impulse deviation is shown in Fig. 3-39 . A variation in thrust (Δ F) just prior to cutoff is assumed to continue on a percentage basis through the entire termination transient, or

$$\Delta I_F = \pm I_N \frac{\Delta F}{F_N} \qquad (1)$$

[redacted]

The time required for the actuators to initiate propellant valve closure is subject to variation ($\Delta \tau$). The effect of this variation on the thrust termination transient is illustrated in Fig.3-39(b). This effect is essentially a delay in the beginning of thrust decay.

$$\Delta I_A = \pm \Delta \tau_A F \tag{2}$$

Main propellant valve closure is also subject to variation. This effect is illustrated by Fig.3-39(c). The relationship between this variation and cutoff impulse will be determined in a later section.

Estimation of Cutoff Impulse Deviation. To provide information concerning the cutoff impulse deviation, estimates were made of cutoff impulse as a function of thrust level and main propellant valve closing time. The prediction of the cutoff impulse of an engine with any accuracy requires a fairly detailed description of the propulsion system. Results of a semiparametric study, such as this, are estimates.

The cutoff impulse was estimated using a transient thrust IBM program developed by Rocketdyne. This program considers the thrust buildup and decay of a propulsion system whose propellant valves are located in the injector. Thus, the effects of combustion during valve closure, and residual gases in the combustion chamber after closure are considered. The effect of residual propellants downstream of the valves (cooling jacket, manifold, etc.) existing in many engines is not considered. This program would approximate an engine with an uncooled thrust chamber or where the main propellant valves are downstream of the cooling jacket. An estimate of the cutoff impulse contribution of the hydrogen in the cooling jacket was made for the J-2. Considering the thermal expansion of the hydrogen which has been heated in the process of cooling the thrust chamber, the estimate indicated that the contribution to the total cutoff impulse was on the order of 8 percent. It should be noted that some of this hydrogen would be burned with residual oxygen left in the manifold, etc at valve closure and the contribution to the cut-off impulse could be somewhat higher. For the storable propellant system this contribution should be negligible.

Using this program, cutoff impulses were determined for thrust levels from 5000 to 100,000 pounds and main valve closing times from 1 to 100 msec. The following assumptions were made:

1.	Propellant combination	LO_2/LH_2	MON/MMH
2.	Chamber pressure	500 psia	200 psia
3.	Mixture ratio	5.0	2.4
4.	L^*	30 in.	40 in.
5.	Both propellant valves close at the same rate		

Results are presented in Fig. 3-40 and 3-41.

Reference 13 presents the following equation for cutoff impulse, assuming instantaneous valve closure.

$$I = \frac{2FL^*}{\sqrt{gk\ RT_{c1}}\ (k+1)} \sqrt{\left(\frac{k+1}{2}\right)^{\frac{k+1}{k-1}}} \qquad (3)$$

[redacted]

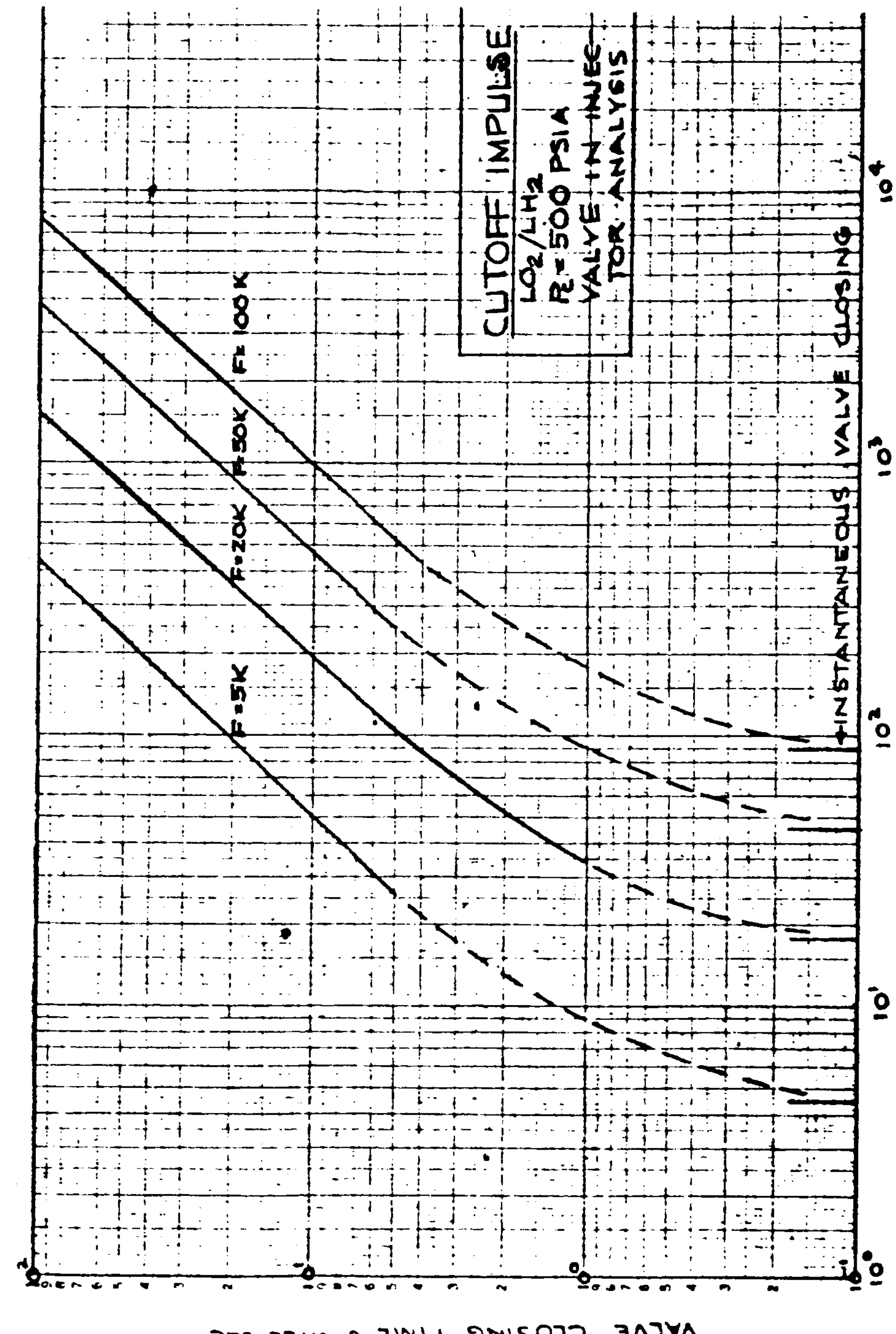


Figure 3-40. Cutoff Impulses vs Valve Closing Time Using LO_2/LH_2

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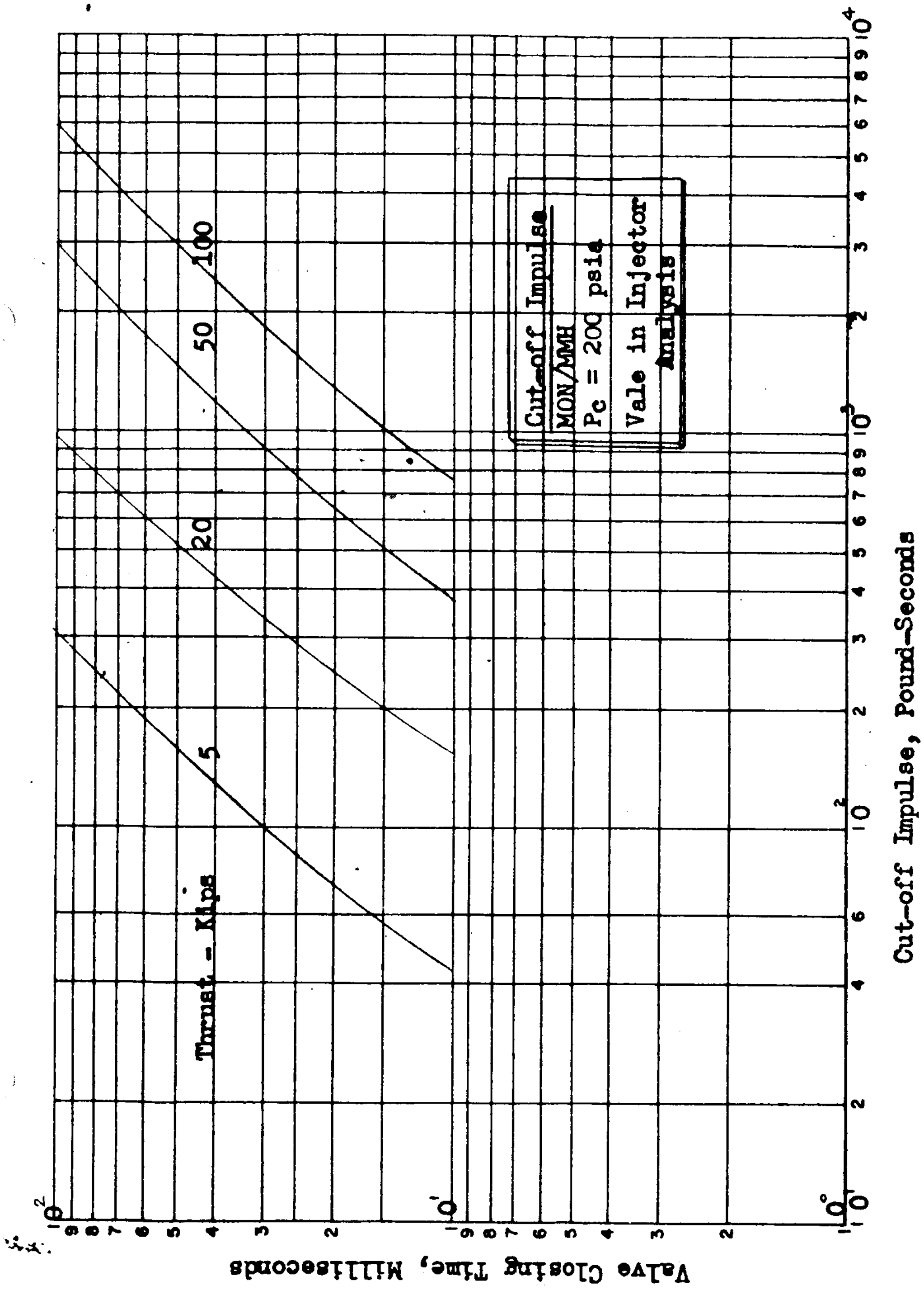

Figure 3-41. Cutoff Impulses vs Valve Closing Time Using MON/MMH

This represents the minimum cut-off impulse and is presented as such in Fig. 3-40. The curves determined for finite valve closure are extrapolated to become asymptotic to these values.

For the straight portion of the curves in Fig. 3-40 and 3-41, the deviation in cutoff impulse due to variation in valve closing time was determined. In the range of thrust and closing time considered, the following relation was found to apply:

$$\frac{\Delta I}{I_N} = 0.90 \frac{\Delta \tau_v}{\tau_v} , \qquad LO_2/LH_2 \tag{4}$$

$$\frac{\Delta I}{I_N} = 0.97 \frac{\Delta \tau_v}{\tau_v} \qquad MON/MMH$$

This is plotted in Fig. 3-42 .

The cutoff impulse deviation due to thrust variation, Eq. 1, is presented in Fig. 3-43. Cutoff impulse variation due to actuator time variation, Eq. 2, is shown in Fig. 3-44.

In the figures above the cutoff impulse deviation is presented in a parametric form. To provide information for the mission studies, assumptions were made as to the variation in the influencing parameters. These assumptions are shown in Table 3-10. The effects of thrust calibration, valve closing time, and actuator time were discussed previously. The variation in propellant density may occur during coasting in space where the propellants are subjected to solar heating, etc. for extended periods of time. This density variation, in turn, affects the engine thrust. A variation in propellant temperature of ± 3 R was assumed

[redacted]

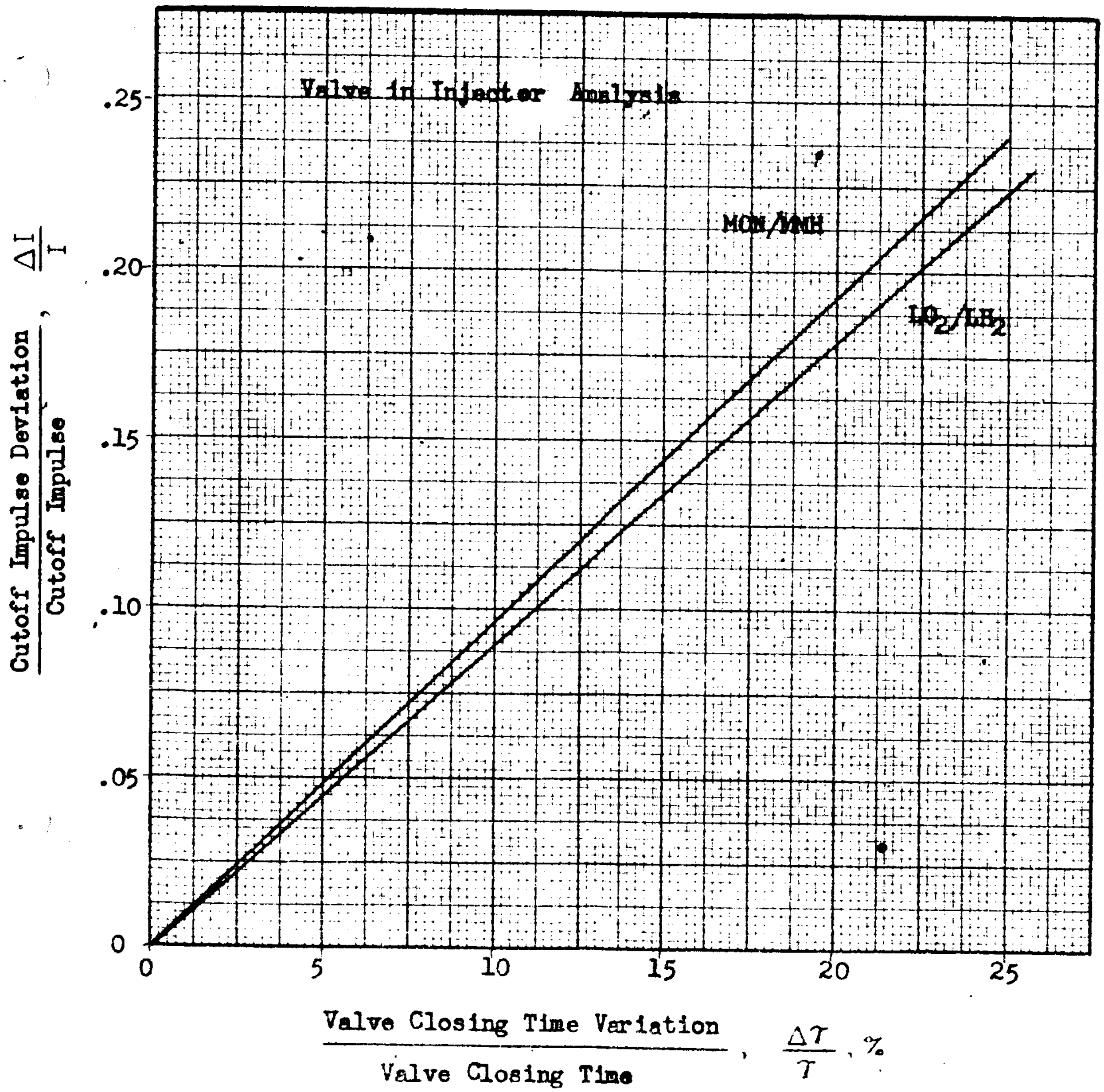

Figure 3-42. Variation in Cutoff Impulse Due to Valve Closing Time Variation

[redacted]

[redacted]

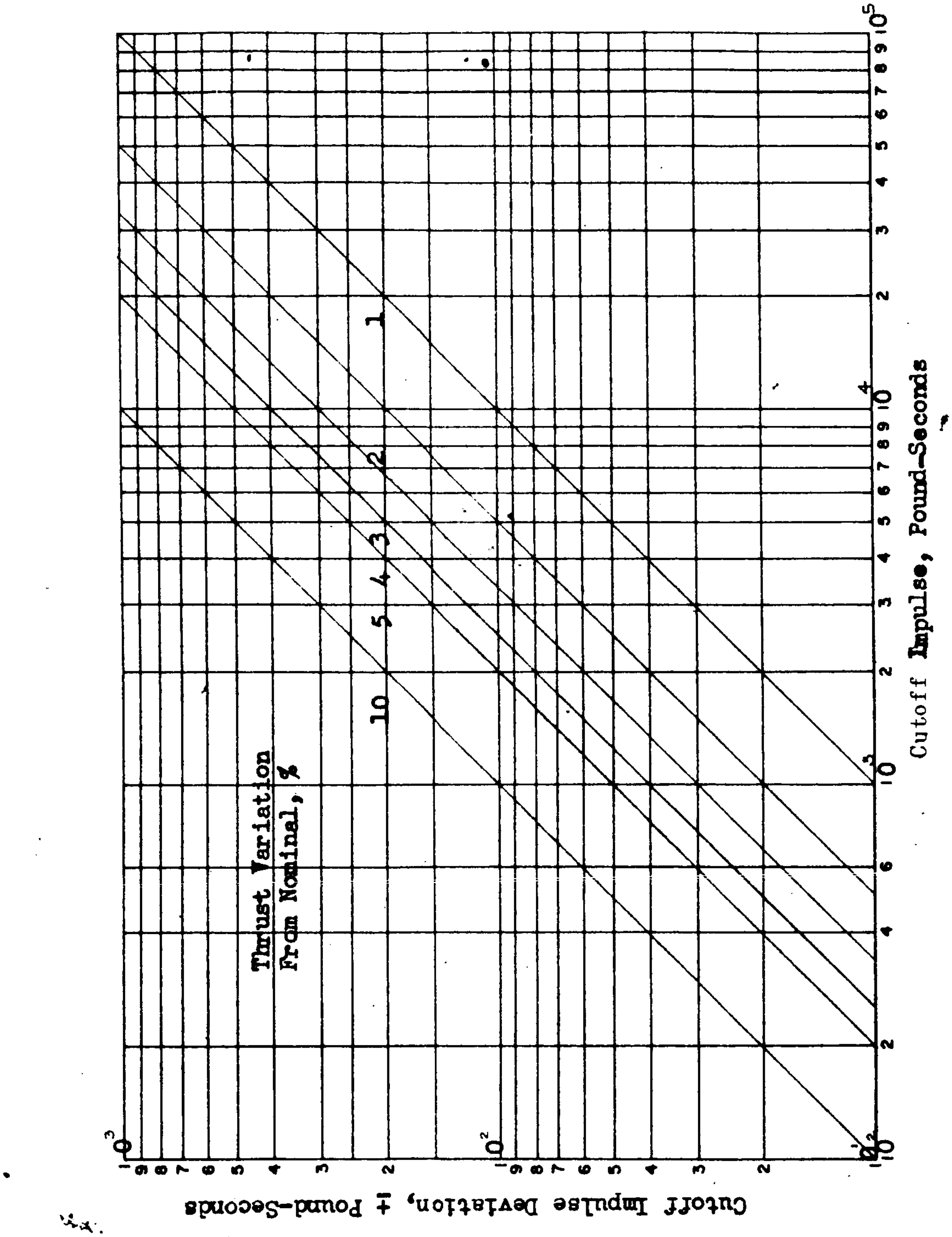

Figure 3-43. Cutoff Impulse Deviation Due to Thrust Variation

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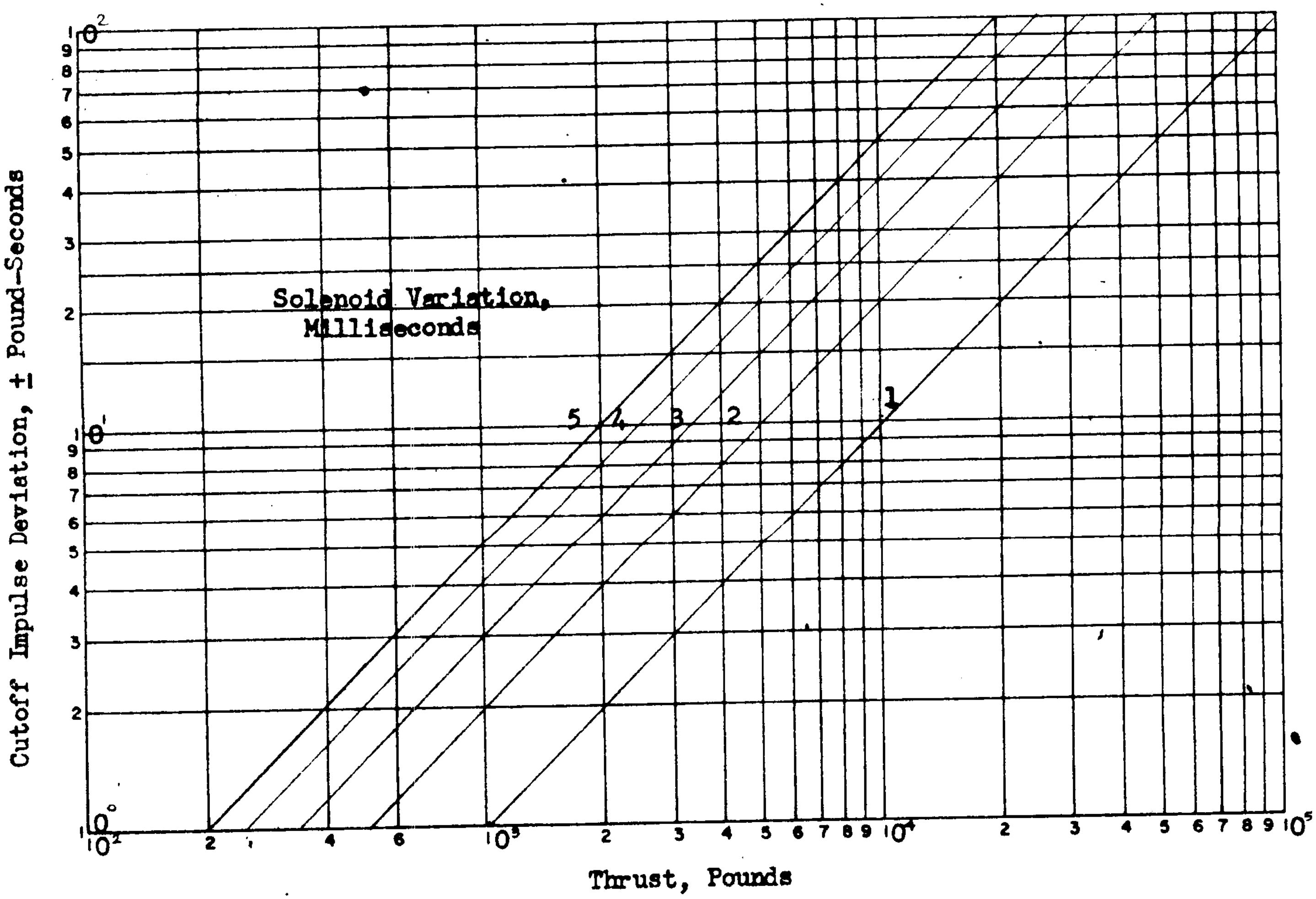

Figure 3-44. Cutoff Impulse Deviation Due to Actuator Variation

TABLE 3-10

ASSUMED VARIATIONS

Variation in Valve Closing Time, percent	± 15
Thrust Calibration (Specific Engine), percent	± 1
Variation in Actuator Time, msec	± 2
Variation in Propellant Density, percent	± 2 (LO_2/LH_2)

for LO_2/LH_2. This effect can be neglected for the storable combination. This results in approximately ± 2 percent variation in propellant density. Through the influence coefficient for an analogous engine, this was related to thrust variation. Using the tolerances assumed, cut-off impulse deviations were determined for a variety of thrust levels and main valve closing times.

Results

For the assumed variations in propulsion system parameters (Table 3-10), the deviations in cutoff impulse are presented in Fig. 3-45 and 3-46 as a function of nominal thrust level and propellant valve closing time. The shaded region represents the valve closing times likely to be encountered.

[redacted]

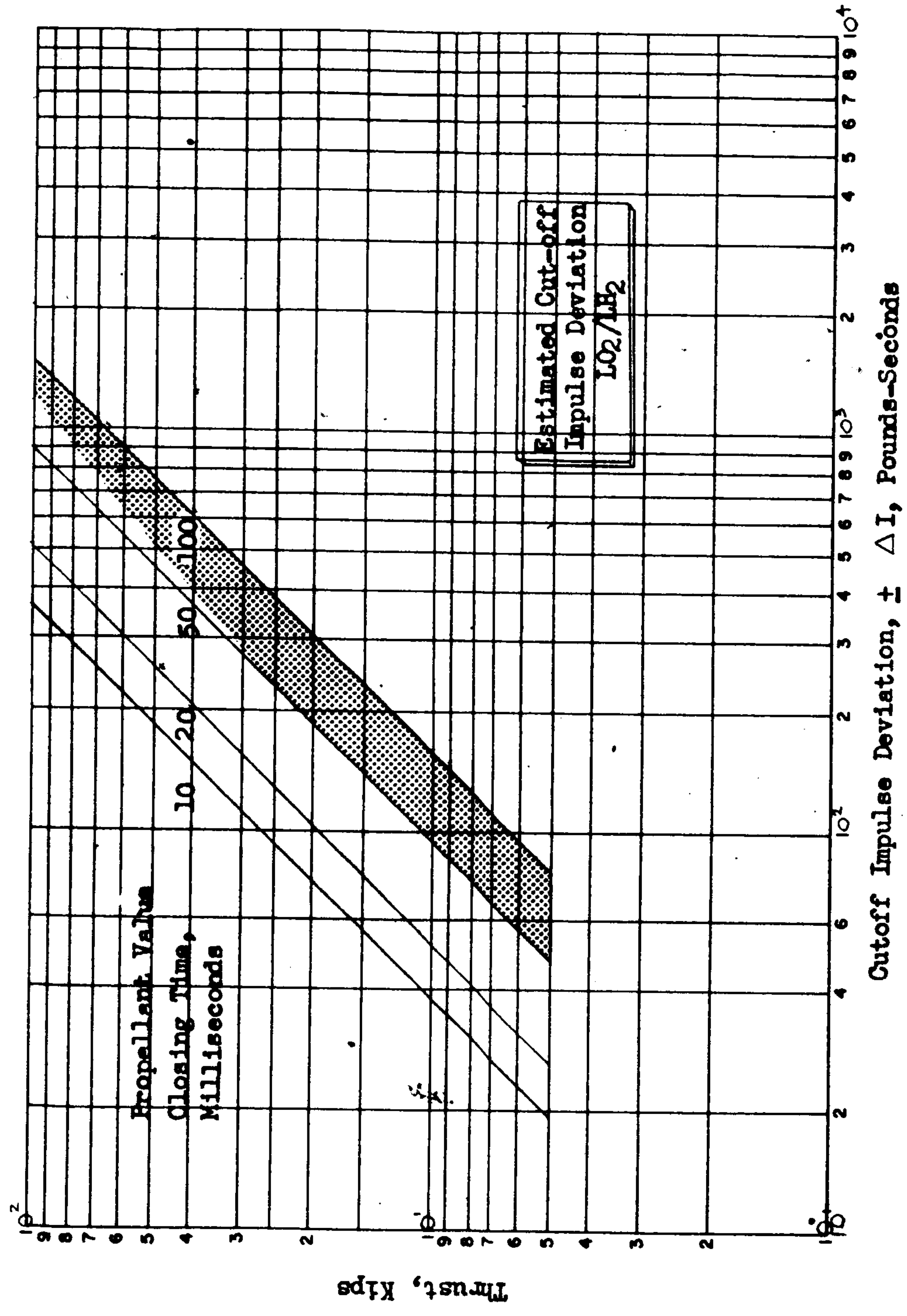

Figure 3-45. Estimated Cutoff Impulse Deviation LO_2/LH_2

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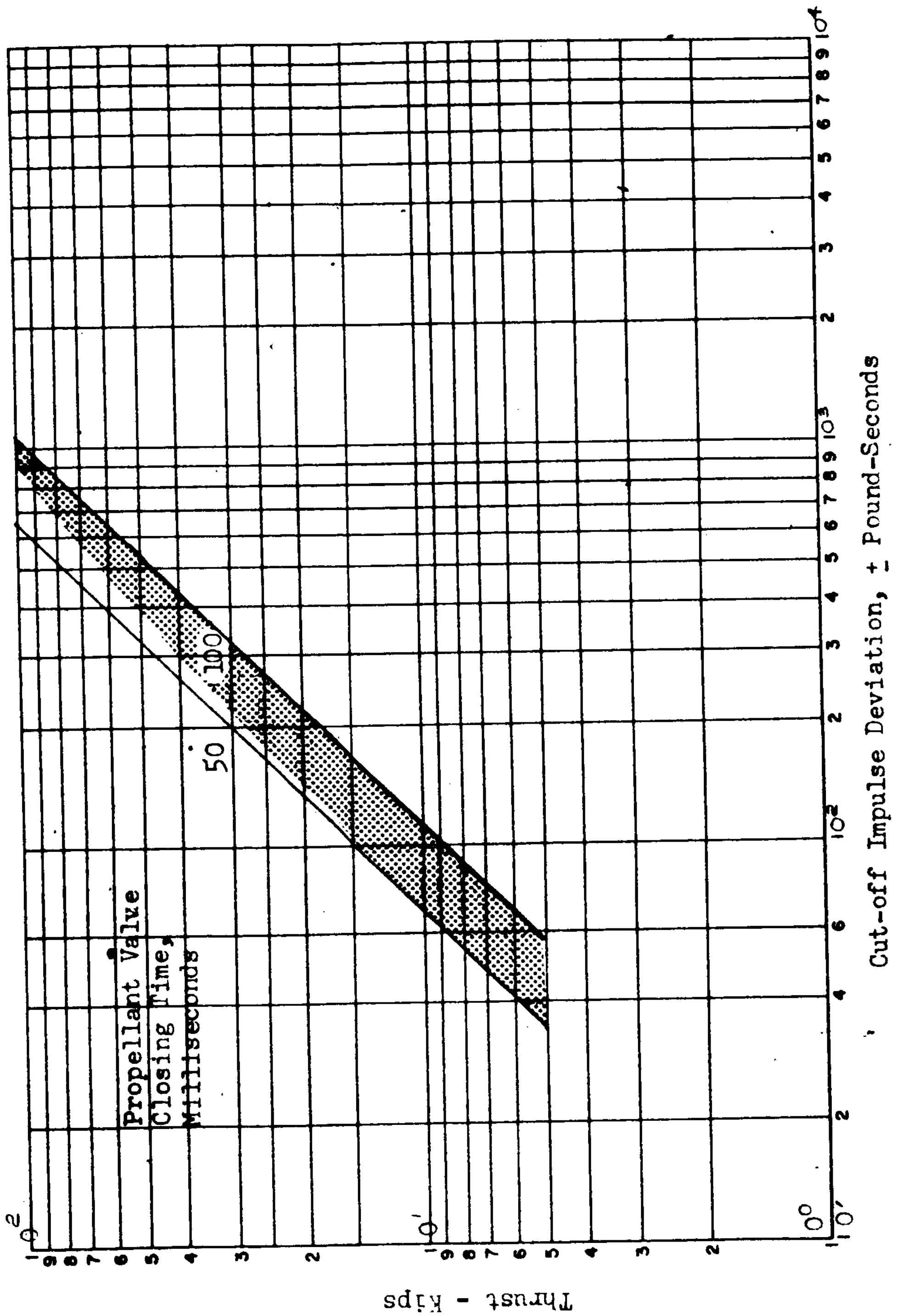

Figure 3-46. Estimated Cutoff Impulse Deviation MON/MMH

These estimates (LO_2/LH_2) seem reasonable when compared to the cutoff impulse deviations predicted for the J-2 if the assumption for variations in main propellant valve closing time are kept in mind. The assumption of this study represents a somewhat more optimistic value than that of the J-2. It is felt, however, that the assumed value is in line with valve performance previously experienced. It should also be noted that use of a propellant utilization system and the resulting mixture ratio variation may cause a considerable deviation in cutoff impulse. Where such a system is used, the propellant utilization valve should be reset to its nominal design position just prior to cutoff.

From the results of this study, the following deviations in cutoff impulse were estimated:

Thrust, 1000 lb	Cutoff Impulse Deviation, lb	
	LO_2/LH_2	MON/MMH
5	± 47	± 35
10	± 110	± 80
20	± 270	± 210
50	± 700	± 570
100	± 1500	± 1110

NOMENCLATURE

I = Cutoff impulse, lb-sec

ΔI = Deviation of cutoff impulse from nominal, 1 lb-sec

F = Thrust, lb

τ = Time, msec

$\Delta\tau$ = Variation in time from nominal value, 1 msec

L^* = Characteristic length, in.

g = 32.2 ft/sec^2

R = Gas constant of combustion gases

T_{cl} = Combustion gas temperature at cutoff initiation, R

k = Specific heat ratio

Subscripts

F = Variation due to thrust

N = Nominal value

A = Variation due to actuator

V = Variation due to main propellant valves

EFFECTS OF THROTTLING ON PROPULSION SYSTEM PERFORMANCE

In certain space maneuvers it may be desirable to use variable thrust (throttled) rocket engines. Missions requiring hovering or rendezvous exhibit this characteristic. Two types of thrust variation may be used: (1) step thrust variation and (2) continuous thrust variation.

Step throttling can be achieved through use of single or multiple chambers. In the multiple chamber case several thrust chambers are used to provide nominal thrust. By cutting the thrust chambers off and on, throttling can be accomplished.

The single chamber step throttling and the continuous throttling are obtained in several ways:

1. Varying the area of the injector orifices
2. Varying throat area
3. Modulating propellant flowrate to result in a lower chamber pressure

The throttling method selected depends upon the purpose and size of the engine system.

This study was conducted to investigate two effects resulting during throttling which may affect the desired mission: (1) degradation of engine performance during throttling and (2) variation of cutoff impulse deviation.

Performance Effects

In general throttling is accomplished by some process which effects a change in chamber pressure (i.e. methods 1 and 3). The vacuum specific impulse obtained during engine operation at a throttled condition may vary from the nominal value in two ways. First the characteristic velocity (c^*) is a function of chamber pressure. Operation at a chamber pressure lower than nominal will result in a decrease in c^* and thus in specific impulse. The second effect would be in the c^* efficiency. Operating a thrust chamber at chamber pressures lower than the design value may cause inefficiencies in operation and thus degradation of performance.

Numerous tests have been made with throttleable engines. Some results from tests on a 150,000-lb, LO_2/RP engine are presented in Fig. 3-47. The change in vacuum specific impulse with throttling is shown in terms of percent of the nominal values. Since the actual tests were made at approximately sea-level conditions, the chamber pressure and characteristic velocity (c^*) were used in transforming this to vacuum conditions. The relation between c^* and chamber pressure based on theoretical combustion performance calculations are also shown.

The data presented in Fig. 3-47 does not reveal any particular relation between throttling and specific impulse. It appears, however, that the c^* efficiency is unaffected by throttling. On this basis it was assumed that the only effect exerted on performance by throttling was the chamber pressure effect on characteristic velocity. This is shown in Fig. 3-48 for the propellant combinations of LO_2/LH_2 and MON/MMH based on an original chamber pressure of 500 psia. It can be seen that the engine can be

[redacted]

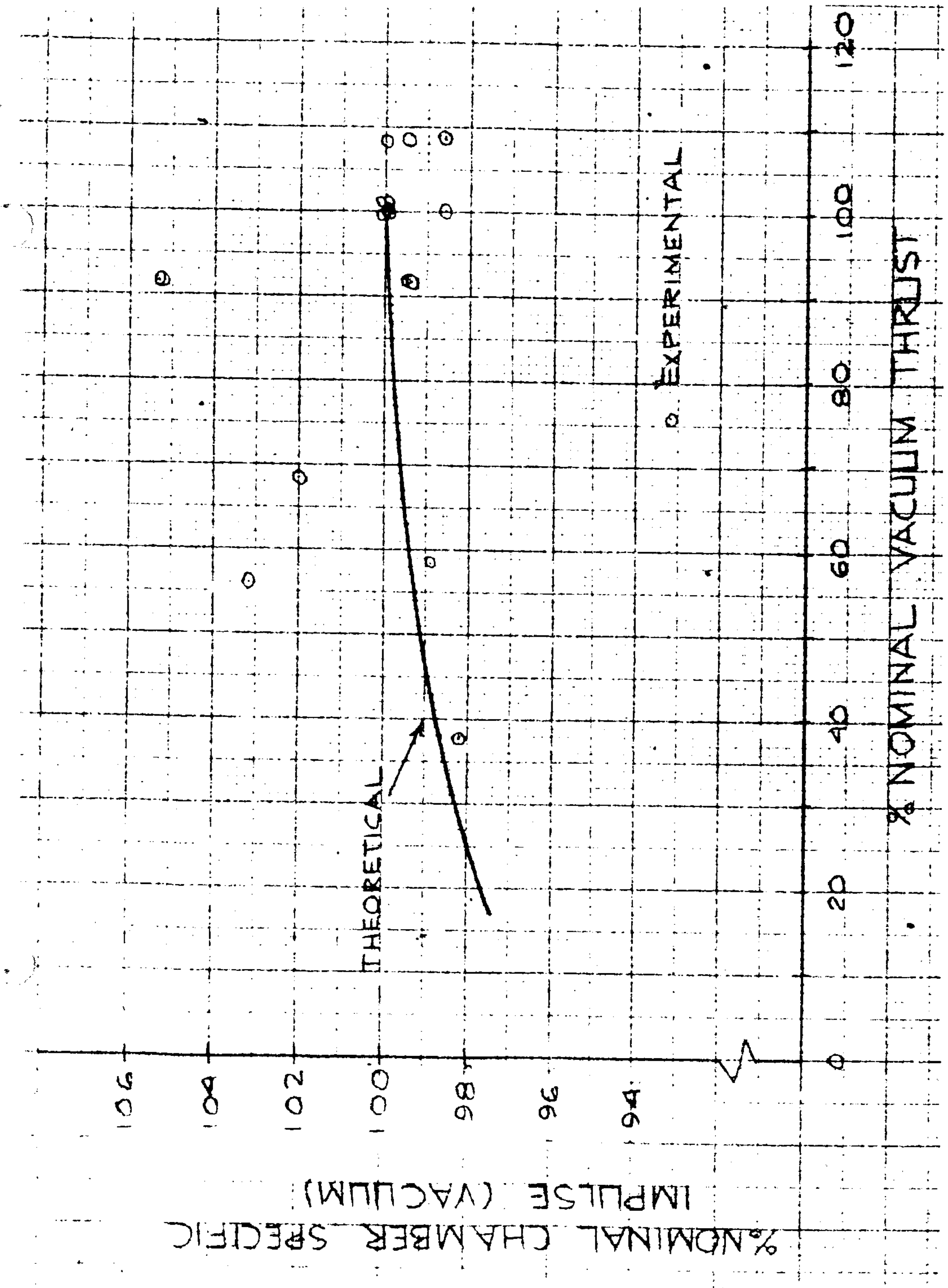

Figure 3-47. Effect of Throttling on Specific Impulse O_2/RP-1

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[redacted]

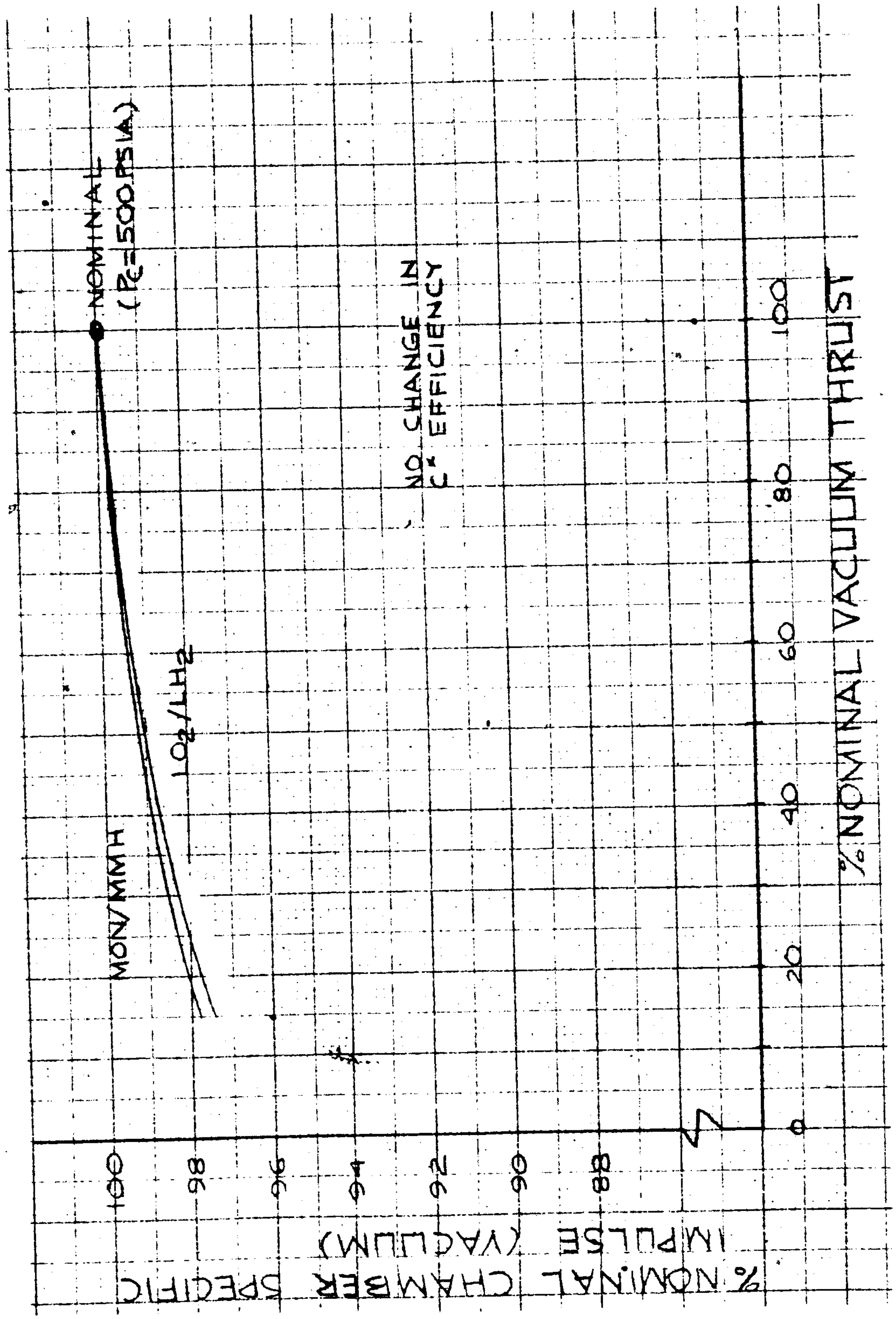

Figure 3-48. Effect of Throttling on Specific Impulse

[redacted]

throttled to some 20 percent of the nominal thrust with only a 2 percent decrease in specific impulse.

Cutoff Impulse Deviation

Previous studies have shown that cutoff impulse deviation is proportional to the thrust of the engine at cutoff. Engines operating at throttled conditions would, therefore, have a cutoff impulse deviation proportional to the thrust level at termination.

During throttling, however, it is anticipated that the mixture ratio cannot be controlled as closely as in the constant thrust engines. A mixture ratio variation of ± 3 percent was assumed. This was assumed to have a proportionate effect on thrust. Other tolerances were the same as those in the cutoff impulse deviation study.

The cutoff impulse deviation for throttled engines is presented in Fig. 3-49 and 3-50 as a function of valve closing time and thrust at engine cutoff. Shaded areas on the figures represent propellant valve closing times that are most likely to be encountered.

RESULTS

From a consideration of the effects of throttling an engine system performance it was found that these effects are slight for engines operating in a vacuum. Specific impulse (thrust chamber) is decreased some 2 percent during 5 to 1 throttling.

[redacted]

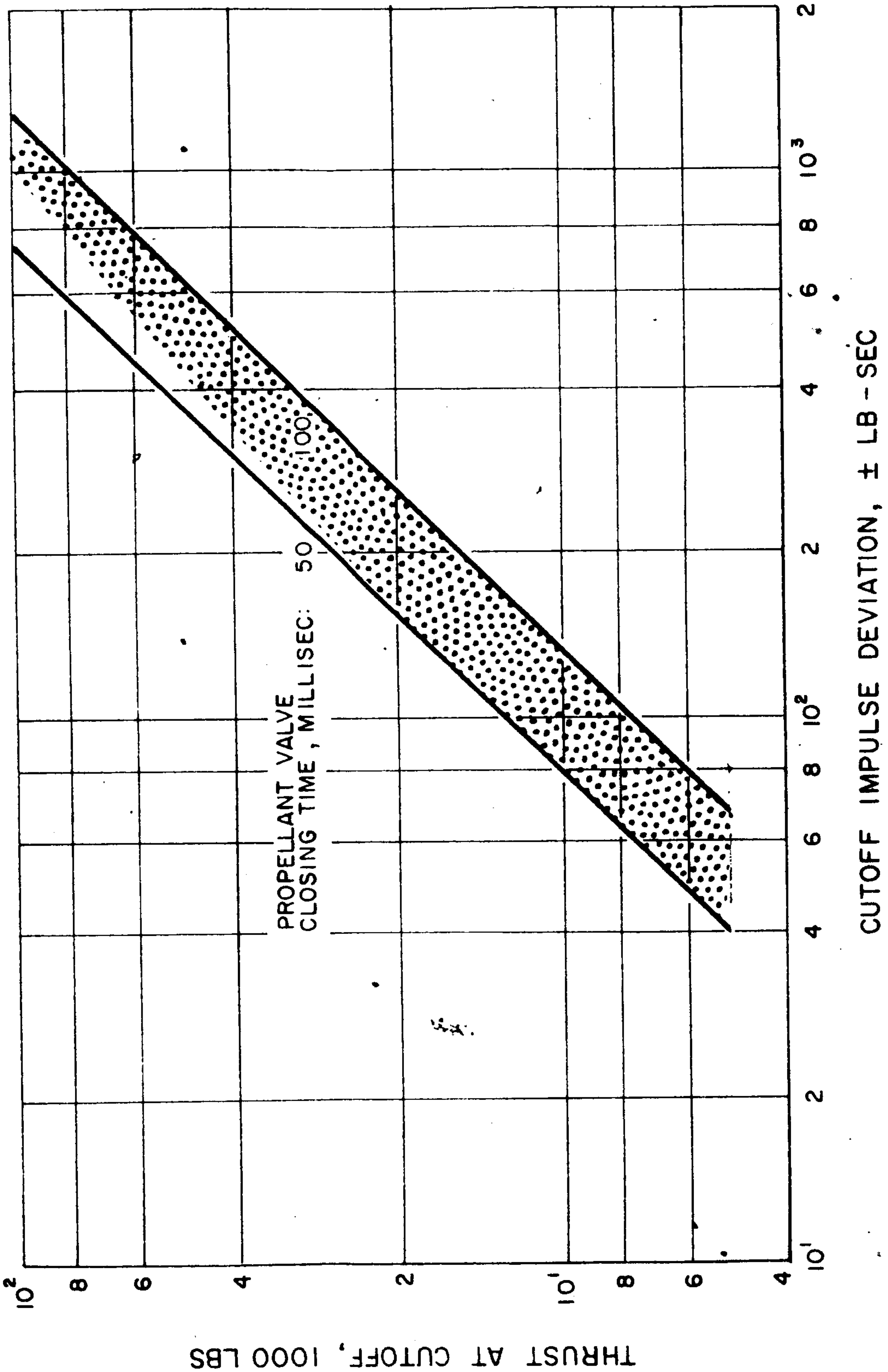

Figure 3-49. Estimated Cutoff Impulse Deviation Throttled Engine MON/MMH

[redacted]

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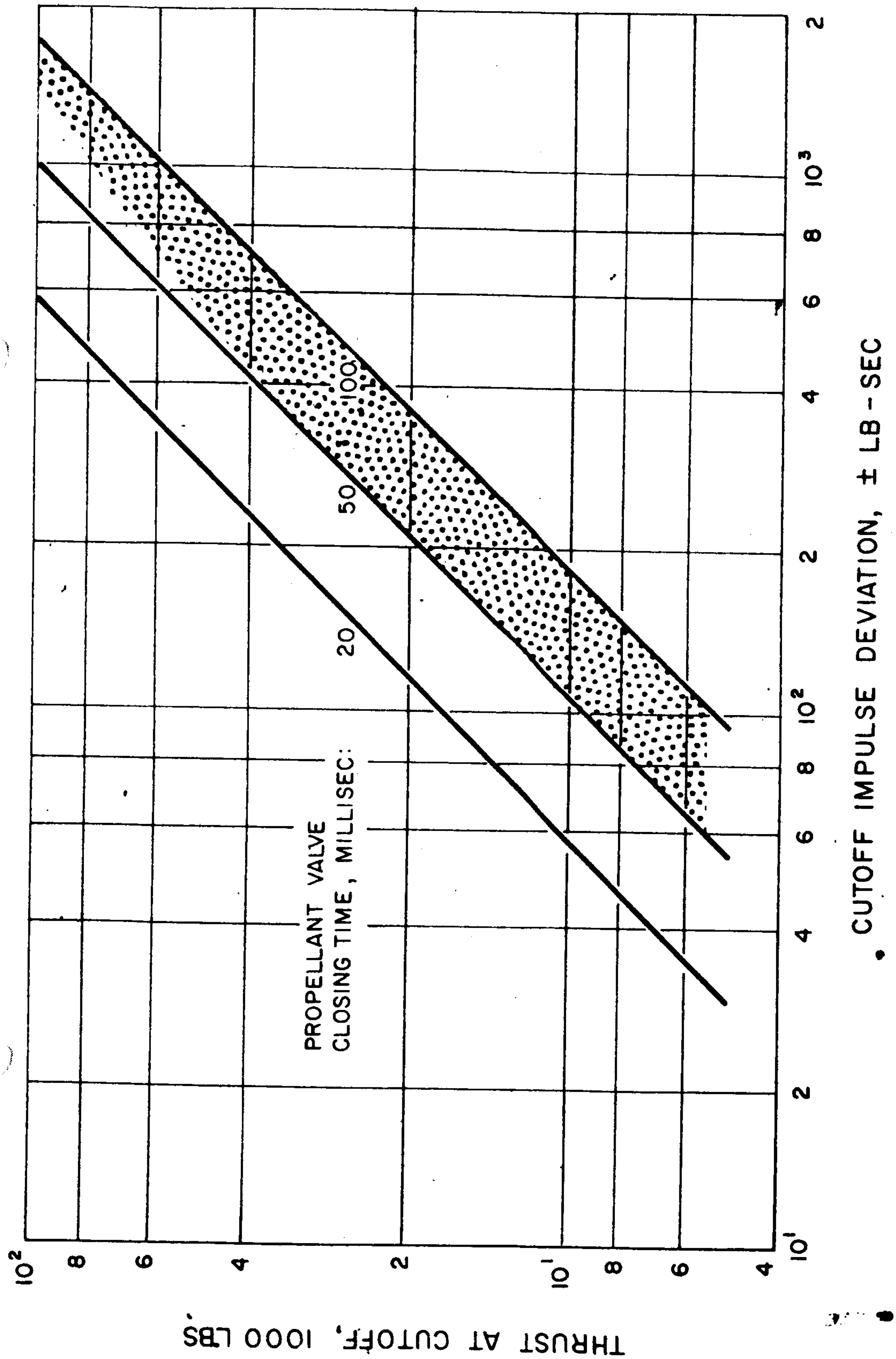

Figure 3-50. Estimated Cutoff Impulse Deviation Throttled Engine LO_2/LH_2

[redacted]

Since cutoff impulse is proportional to thrust, an engine operating at a throttled condition will have a correspondingly lower cutoff impulse deviation.

PROPULSION SYSTEM DESCRIPTION

For the space vehicles four basic propulsion system configurations were considered. These are presented along with their engine operating parameters in Table 3-11. The pump fed engine specific impulses have been reduced to account for gas generator flowrate. These propulsion systems were used in the various mission studies that were conducted.

TABLE 3-11

PROPULSION SYSTEM PARAMETERS

Propellant	Feed System	P_c	ϵ	o/f	I_s
LO_2/LH_2	Pump	500 psia	30	5:1	428.8
LO_2/LH_2	Pressure	60 psia	16	5.5:1	401
MON/MMH	Pump	500 psia	30	2.4:1	323.4
MON/MMH	Pressure	150 psia	25	2.4:1	320

These parameters were determined from considerations discussed in section 5 . The weights for these systems have been identified in two categories: those which are propellant dependent and those which are thrust dependent. Table 3-12 shows the weight breakdown.

Some of the propellant dependent weights are the same for all four systems. Propellant and gas lines, ducts and valves (not a part of the engine system), interstage structure, boattail, attach members, etc. have been lumped into a miscellaneous propellant dependent term which is 1.5 percent of the usable propellant weight. Unusable residual propellants

[redacted]

TABLE 3-12

WEIGHTS

Propellants	LH_2/LO_2		MON/MMH	
Feed	Pump	Pressure	Pump	Pressure
<u>Propellant Dependent Weights</u>				
1. Propellant Tank	0.0165 Wp	0.0428 Wp		0.0289 Wp
(Minimum Thickness)	$\left[0.481\ (Wp)^{2/3}\right]$		$\left[0.199(Wp)^{2/3}\right]$	
2. Pressurizing System	0.0040 Wp	0.0348 Wp	0.0017 Wp	0.0174 Wp
3. Lines, Valves and Miscellaneous	0.015 Wp	0.015 Wp	0.015 Wp	0.015 Wp
4. Residual Propellants	0.010 Wp	0.010 Wp	0.010 Wp	0.010 Wp
5. Flight Performance Reserve	0.010 Wp	0.010 Wp	0.010 Wp	0.010 Wp
6. Meteoroid Shielding	$0.477\ (Wp)^{2/3}$	$0.460\ (Wp)^{2/3}$	$0.219(Wp)^{2/3}$	$0.219(Wp)^{2/3}$
7. Insulation (SI-4):				
Lunar 3/4 in.	$0.300\ (Wp)^{2/3}$	$0.290\ (Wp)^{2/3}$		
Mars 3 in.	$\left[1.200\ (Wp)^{2/3}\right]$			
Propellant Dependent Weight	0.0555 Wp $+0.777\ (Wp)^{2/3}$	0.1126 Wp $+0.75\ (Wp)^{2/3}$		0.0813 Wp $+0.219(Wp)^{2/3}$
(Minimum Thickness)	$\left[0.039\ Wp + 1.258\ (Wp)^{2/3}\right]$		$\left[0.0367\ Wp + 0.418(Wp)^{2/3}\right]$	

[redacted]

TABLE 3-12

(Continued)

Propellants	LH_2/LO_2		MON/MMH	
Feed	Pump	Pressure	Pump	Pressure
Thrust Dependent Weights				
1. Thrust Chamber, lb/1000-lb	4.40	25.24	4.73	16.30
2. Turbopump, lb/1000-lb	3.02		2.33	
3. Miscellaneous, lb/1000-lb	2.93	3.00	2.93	3.00
Thrust Dependent Weight, lb/1000-lb	10.35	28.24	9.99	19.30

are assumed to be 1 percent of the usable propellant. An additional 1 percent of the usable propellant is loaded as flight performance reserve, to allow for variations in thrust and specific impulse. As described in the environmental study, meteoroid shielding can be provided by a 0.032-in. aluminum sheet surrounding the entire stage. The weight of this shield is proportional to the surface area and is therefore proportional to the propellant weight to the 2/3 power. The environmental studies indicate that the LO_2/LH_2 stages will require insulation to eliminate propellant loss due to boiloff. The insulation selected is "Linde SI-4." This insulation thickness was 3/4 in. for all LO_2/LH_2 stages for the lunar mission and for the first stage of the Mars mission. For subsequent stages of the Mars vehicle, where long storage times were required, the insulation thickness was increased to 3 in. The insulation weight is proportional to the propellant weight to the 2/3 power since it is wrapped around the entire tank.

The weight of the tanks, pressurizing subsystem, and engine subsystem are dependent on individual system design. The propellant tank weight includes the tank skin plus an increase of 20 percent for the pump-fed system and 10 percent for the pressure-fed systems to allow for skin thickness variations and welds. The thrust dependent weights are divided into three parts: thrust chamber, turbopump, and miscellaneous weights.

LO_2/LH_2: Pump Fed

The tank configuration for this system consists of a spherical LO_2 tank topped by a cylindrical hydrogen tank with hemispherical ends. A common bulkhead was not used between tanks since this would increase the heat transfer between propellants. The tank design pressure for this system was 25 psia and a 5 percent ullage was allowed in determining the tank volume to propellant weight relationship. With this low tank pressure the tank will reach the minimum thickness of 0.032 in. when the propellant weight is below approximately 25,000 lb. The pressurizing subsystem consists of a dual heat exchanger, with each propellant vaporized to pressurize its own tank. The engine weights for this system are in agreement with an extrapolation of the J-2 weights.

LO_2/LH_2: Pressure Fed

The propellant tanks for this system are similar to those for the LO_2/LH_2 pump-fed system except that the tank design pressure is 150 psia. A 5-percent ullage space is allowed in each tank. The pressurizing subsystem consists of helium gas stored at 4500 psia in fiberglass spheres. To reduce the weight of the propellant tanks and the pressurization system a mixture ratio of 5.5 to 1 was selected.

The low chamber pressure of this system results in a heavy thrust chamber, but lighter propellant tanks and pressurization subsystems.

MON/MMH: Pump Fed

The propellant tanks for this system have a common bulkhead since there is no heat transfer problem. The fuel tank is cylindrical with a conical bottom and an inverted hemispherical top which is also the bottom of the oxidizer tank. The cylindrical oxidizer tank has a hemispherical top. The tank design pressure is 25 psia and 3 percent ullage space was provided. The tanks are minimum thickness (0.032 in.) for most propellant weights considered so the tank weight is proportional to the propellant weight to the 2/3 power. The pressurization subsystem considered is a dual gas generator system. The gas generator to pressurize the fuel tank operates fuel rich, while the gas generator to pressurize the oxidizer tank operates oxidizer rich. This thrust chamber weighs slightly more than the LO_2/LH_2 chamber because of the more dense propellants which are trapped in the chamber, but the turbopump is lighter because of the reduced pumping head required.

MON/MMH: Pressure Fed

The tank and pressurizing subsystems for this storable pressure-fed system are the same as the storable pump fed system except that the tank pressure is 225 psia. Since the propellant density for this engine is higher than for LO_2/LH_2, the tank weight will be lower, even though a higher tank pressure and chamber pressure are used. This higher chamber pressure results in this engine being considerably lighter than the LO_2/LH_2 pressure fed engine.

SECONDARY PROPULSION SYSTEM CONSIDERATIONS

In addition to the topics discussed which significantly affect the propulsion system there are some secondary aspects which should be considered. Trapped propellant, use of a propellant utilization system, thrust vector control requirements, and acceleration loads all affect the propulsion system design. These features are discussed in the following sections.

Trapped Propellant and Propellant Utilization Systems

The purpose of this study was to investigate the amount of propellant that may be trapped in a space vehicle propulsion system and indicate the advisability of using a propellant utilization system. Trapped propellant as it is used in this study means that propellant which is left unburned due to premature exhaustion of the other propellant. As such, the definition excludes residual propellants which are trapped in lines and engine, used in engine start, and flight performance reserves in which the system is deliberately overtanked to provide a propellant allowance for possible specific impulse variation. The trapped propellant was evaluated using a statistical analytical technique whereby an optimum fuel bias was selected which maximized the probability that the trapped propellant does not exceed a specified acceptable value. The effect of this trapped propellant on the performance of a propulsion system was evaluated, and the desirability of using a propellant utilization system discussed. The system considered uses liquid oxygen/liquid hydrogen and is pump-fed.

Analysis. Trapped propellant at the end of a missile flight is a result of (1) off mixture ratio tanking, and (2) deviation from the expected time-average mixture ratio operation. The off mixture ratio tanking is due to inaccuracies in determining the weight of propellant that has been tanked. Deviation from expected time-average mixture ratio operation can be attributed to fluctuations in (1) propellant density, (2) pressure regulator repeatability, (3) instrumentation and data reduction accuracies, and (4) missile acceleration history.

Propellant Weight Variation. The loaded propellant weight variation could be substantial considering (1) propellants are loaded at the launch site, (2) both propellants are cryogenic, and (3) propellants must be maintained in the tanks during what may be lengthy countdowns. Space missions, however, will be of an individual nature and considerably more care could be exercised in loading the propellant. From these considerations, and information accumulated from the tanking accuracy of various missiles, it was assumed that propellant weight estimates would be accurate to 1 percent. This was considered to be a 2σ value.

Mixture Ratio Variation. Assuming that the fluctuations leading to mixture ratio variation are independent and normally distributed, it is possible to arrive at some distribution around the expected time average mixture ratio. The fluctuation of these factors is discussed and the standard deviation (2σ) calculated.

Propellant Density. It is assumed that in spite of the effects of environment and storage time on these cryogenic propellants, density can be predicted to ±2 percent. Through the use of influence coefficients the effect of this variation on mixture ratio was found to be ±2.6 percent.

Pressure Regulator Tolerance. The pressure regulator tolerance was assumed to be ±5 percent of the tank operating pressure. This results in about ±1.5 psi variation in both the oxidizer and fuel tanks. Using the influence coefficients the mixture ratio variation was found to be ±0.2 percent.

Instrumentation and Data Reduction Inaccuracies. During the acceptance testing of a propulsion system, data regarding performance are recorded, reduced to standard conditions and the engine accepted or rejected on the basis of these static tests. Assuming the missile receiving any propulsion system is tanked utilizing information gained during acceptance testing, the mixture ratio may deviate due to inherent testing inaccuracies. A mixture ratio variation of ±0.5 percent is assumed based on current engine acceptance programs and represents a 95 percent degree of confidence.

Missile Acceleration History. As a missile accelerates and ascends, the propulsion system performance (thrust, mixture ratio, flowrate etc.) is constantly changing due to drag variations, increased system pressures and decreased atmospheric pressures.

During the preliminary design of a missile, estimates of all the above parameters are made in accord with the best available state-of-the-art information. These estimates are then fed into various optimization programs, and result in information such as acceleration, thrust, and specific impulse as a function of time from launch. Once hardware is fabricated, these estimates (i.e. component weights, engine performance) are revised to the values realized during the development phase of the program. One can see, then, that the accuracy of the output from the above mentioned optimization programs is a function of development time.

The shape of the acceleration vs time curve is strongly a function of takeoff thrust and takeoff missile gross weight. (It is insensitive to the trajectory deviations considering reasonable perturbations around the optimum.)

The missile thrust and gross weight are assumed to vary ±3 percent each, resulting in a ±6 percent variation in acceleration. Use of the influence coefficients results in a mixture ratio variation (95 percent confidence, 2σ) of ±0.3 percent.

Standard Deviation. The standard deviation is found from these individual deviations as previously mentioned. This standard deviation was found to be ±1.51 percent. These values are used in plotting the curve of Fig. 3-51.

It should be noted that certain factors in addition to those mentioned may have an effect on mixture ratio. These factors, (1) variation in launching altitude, (2) tank dimensional tolerances, and (3) variation in propellant consumption prior to full thrust, were assumed negligible in this preliminary study.

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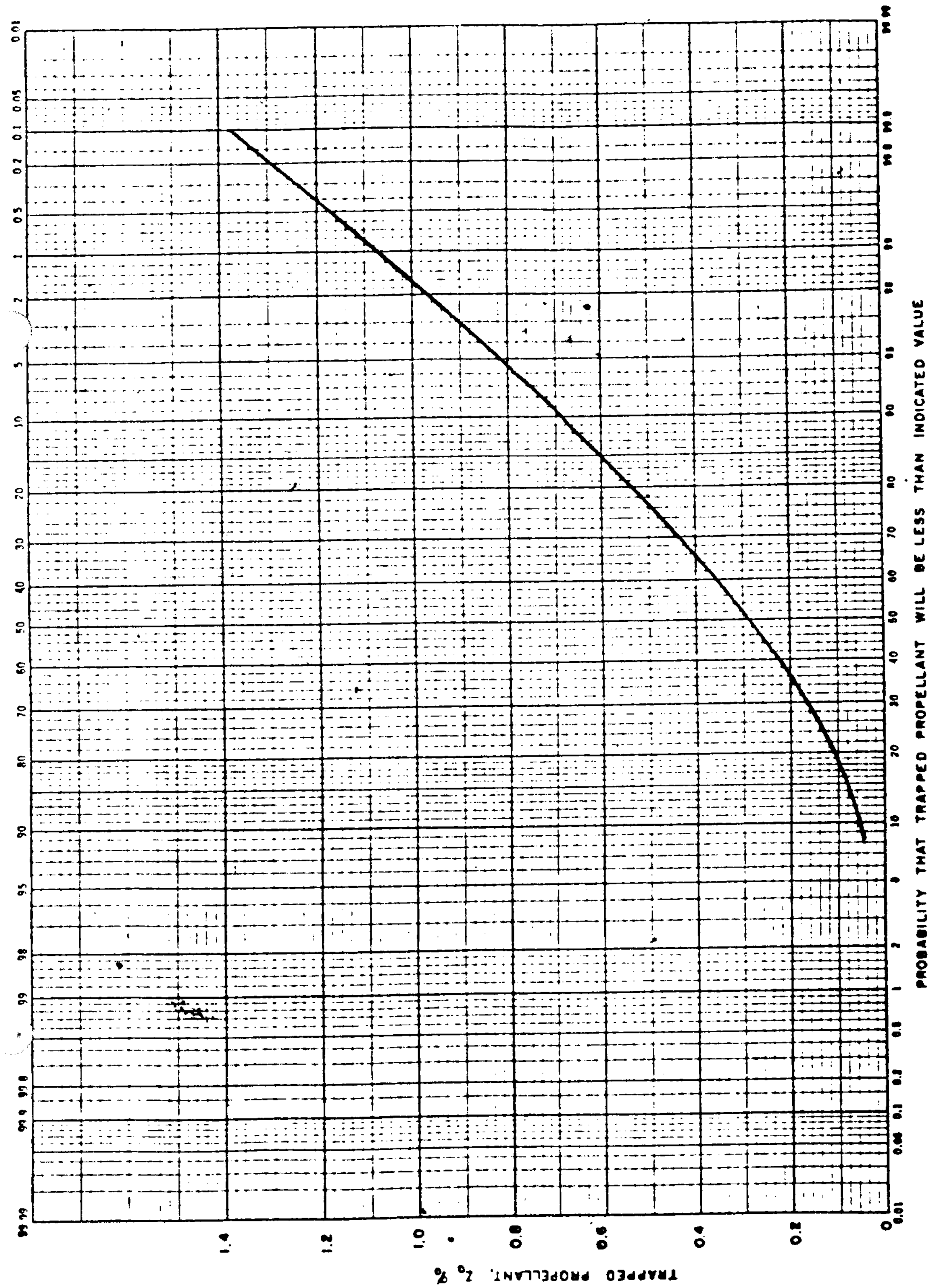

Figure 3-51. Trapped Propellant Probability

[redacted]

The curve of Fig. 3-51 shows the probability of the trapped propellant being less than a certain value (Z_0). It can be seen that there is a 99.9 percent probability that the trapped propellant will be less than 1.38 percent.

Effect of Trapped Propellant on System Performance. The effects of trapped propellant on propulsion system performance are studied by considering the effect on the payload-to-gross weight ratio of a single stage LO_2/LH_2, pump-fed vehicle. For a given ideal velocity requirement (ΔV) the payload-to-gross weight ratio can be evaluated as a function of the percent of trapped propellant. This in turn can be related to its probability of occurrence through Fig. 3-51.

The results of these calculations are presented in Fig. 3-52 for 1.38 percent trapped propellant which corresponds to 99.9 percent probability. It can be seen that for low velocity requirements the trapped propellant has slight effect on the payload. For higher velocity requirements the effect is more pronounced.

Propellant Utilization System Considerations. The effect of a propellant utilization system should now be considered. Basically, a propellant utilization system can be expected to decrease the variance of mixture ratio. A propellant utilization system, then, tends to compress the mixture ratio distribution.

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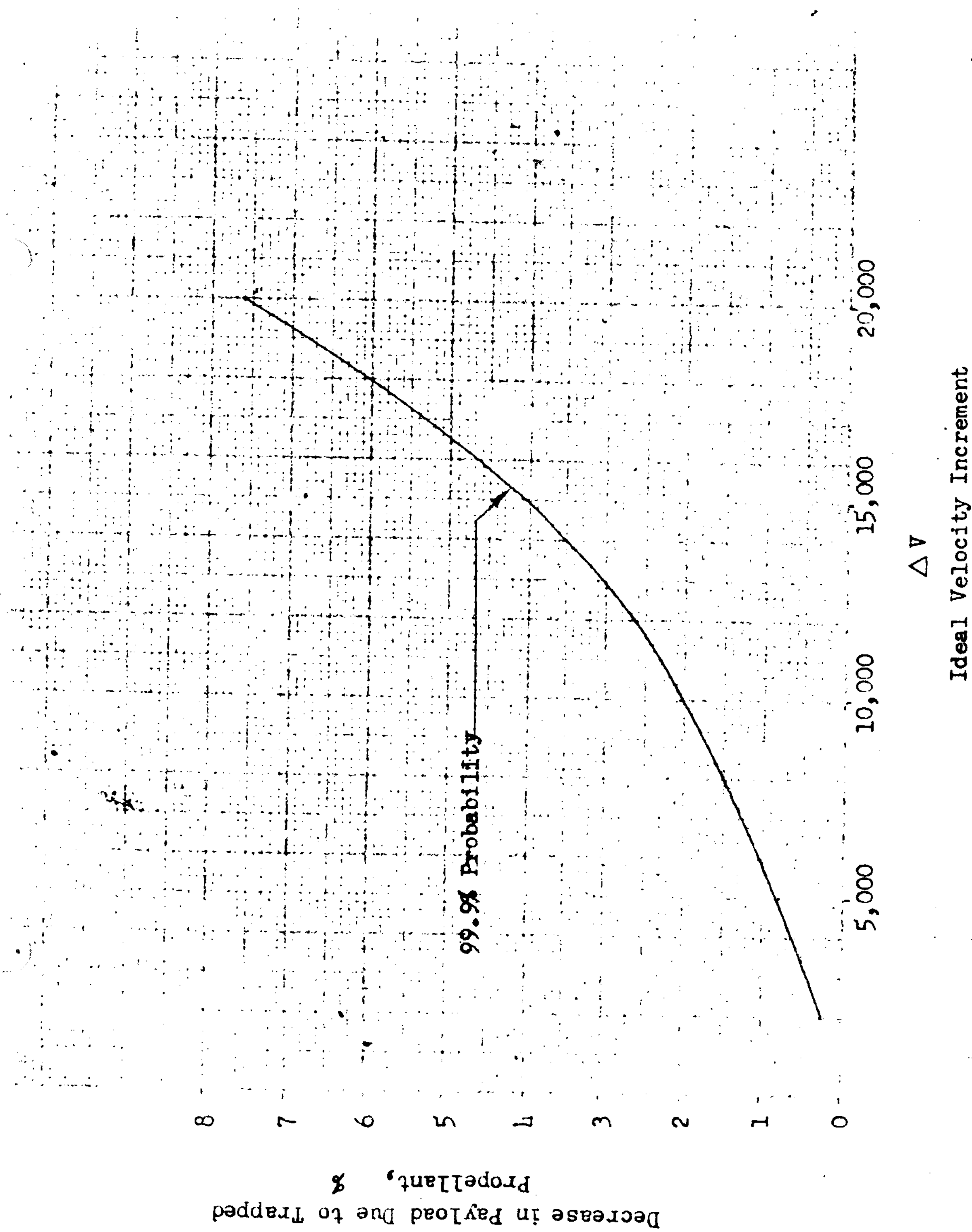

Figure 3-52. Effect of Trapped Propellant on Engine System Performance

[redacted]

Any propellant utilization system will have certain disadvantages. These are listed below:

1. Weight

   a. The propellant utilization system will consist of certain control and sensing components. The weight of these components must be subtracted from the payload advantage of a missile stage with a propellant utilization system to obtain the true payload advantage.

2. Reliability

   a. Because of the added complexity of a propellant utilization system, the over-all reliability of an engine with a propellant utilization system will be lower than that of an engine without a propellant utilization system.

3. Development Time and Cost

   a. The cost of a propellant utilization system can be a significant addition to the basic engine development cost.

One additional difficulty which could considerably complicate development of a propellant utilization system is the low density and cryogenic nature of the hydrogen fuel. The consequent lack of a well-defined liquid surface and the small pressure differences existing in the fuel tank raise doubts about the attainable accuracy with any tank sensing devices. A flow integration system would also be subject to large fuel density variations during a mission.

From consideration of the complexities of a propellant utilization system and the deleterious effects of trapped propellants the following general remarks can be made: (1) for low energy (low ΔV) missions where the effects of trapped propellant are not significant a propellant utilization system would not be warranted, and (2) in high energy missions a propellant utilization system can be used to good advantage.

<u>Thrust Vector Control</u>

A preliminary review has been made of the thrust vector control requirements for space stage pitch, yaw and roll axis orientation. Requirements for thrust vector control will result from three principle sources:

1. Thrust vector control to correct for engine (and vehicle) misalignment
2. Thrust vector control to correctly reorient the space stage after separation, assuming a torque existed at separation from a nonuniform (unsymmetrical) previous stage shutdown
3. Thrust vector control to maintain the correct vehicle attitude during space stage operation

Basic analyses have been reviewed from presently available data to indicate possible requirements. In regard to the vehicle center of gravity location, mass moment of inertia, and weight, the values used in the calculations are based on assumed design criteria that may differ from any final space stage design. However, the thrust vector control results obtained will indicate trends in the requirements.

The difficult problem that arises in selecting thrust vector control requirements is the margin of overdesign, or margin of safety that should be added. Reviewing past engine designs (both booster-and upper-stage systems) it can be shown from test flight data that the actual engine gimbaling is much less than the maximum for which the engine is designed.

While booster and upper stage engines have design gimbal angles of approximately 5 to 10 deg, it is very difficult to find an actual gimbal angle in flight approaching a magnitude of 1 deg.

The original design basis for the booster and upper stage engines is usually a simulated flight trajectory with a maximum wind profile. As a result, when vehicles are not flight tested during severe winds, the large gimbal angles are not required. For an upper stage engine which functions outside the Earth's atmosphere, the required gimbal angle from a simulated trajectory to provide the preprogramed vehicle attitude is very small. The margin of overdesign required for maximum wind loading in a booster stage engine is not required in such upper-stage engines.

Thus the calculations that have been generated regarding thrust vector control requirements must be viewed in the light that a large margin of safety has historically been included in past engine specifications.

<u>Thrust Vector Requirements for Misalignment</u>. Angular or lateral engine thrust vector misalignment with respect to the vehicle center of gravity will cause a torque about the vehicle center of gravity. Based on previous engine manufacturing tolerances the engine thrust vector misalignment should not be greater than ±0.25 in. lateral, and ±0.5 deg angular.

The tank and structure may have a lateral center of gravity misalignment resulting from tank manufacturing tolerances, tank bending, or unsymmetrically attached components. A noncircular, nonsymmetrical tank shape (such as a pear-shape cross section), where the lateral center of gravity would not be in the predicted longitudinal axis, may be used. Discussions with the vehicle contractor personnel, involved in tank design for a variety of presently operating large vehicles, have indicated that small lateral misalignment values occurring from tank asymmetry are insignificant to engine gimbal requirements. Nonsymmetrically located components cause a lateral center-of-gravity shift as the propellant is consumed. This analysis has assumed that symmetrical component placement can exist and that the vehicle tank lateral misalignment due to noncircular shape is no greater than ±0.25 in.

<u>Pitch and Yaw</u>. Misalignment of the main thrust chamber thrust vector will produce a lateral torque about the vehicle center of gravity in either the pitch or yaw axis, or a combination of both.

For a vehicle considered, employing an 85,000-lb-thrust engine, a graphical presentation of the upsetting torque (for the burnout condition which is the worst) resulting from a range of lateral and angular misalignment was formulated (Fig. 3.53). For the total assumed misalignment, the upsetting torque resulting would be approximately 50,000 lb-ft.

This upsetting torque would require correction either from an auxiliary attitude control system, or gimbaling the main engine so the upsetting torque is not produced. A maximum engine gimbal angle of 0.65 deg would be required.

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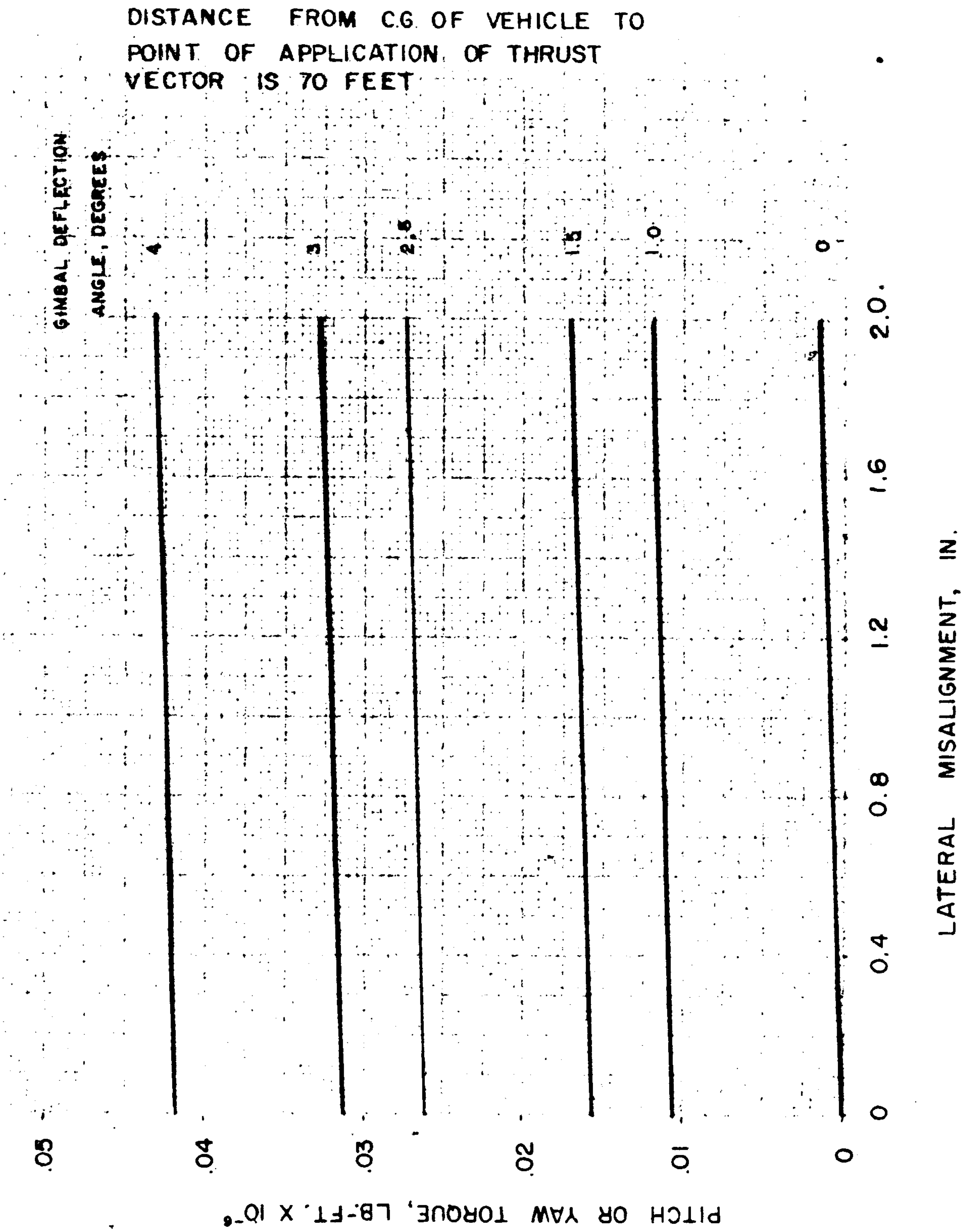

Figure 3-53. Upsetting Torque

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[redacted]

Roll. A lateral misalignment in combination with an angular misalignment (or engine gimbaling) will produce a roll torque. However, this resultant roll torque is low. With the above mentioned vehicle and angular misalignment values the roll torque produced is 30 lb ft; for a lateral misalignment of 0.5 in. and an engine gimbal angle of 3 deg the roll torque produced is 185 lb ft.

Torque From Previous Stage Shutdown. The transition conditions in an engine shutdown produce a cutoff impulse which may have a variation. In a vehicle stage which has a multiple engine system, the variation in cutoff impulse between the engines will produce a torque in either the pitch or yaw axis, or a combination of both. This torque will produce a vehicle angular rate which will be inherited by the following stage after separation. Thus a control torque in the following stage is necessary after separation to correctly reorient the vehicle attitude.

An analysis was conducted based on a preceding stage with four J-2 engines. A statistical analysis of the torque impulse imparted to a space vehicle at second-stage shutdown indicates that for assumed standard misalignments and cutoff impulse deviations of each of the four 200,000-lb-thrust J-2 engines, there is a 99.7 percent probability that the cutoff torque impulse would be not greater than 117,000 lb ft-sec in any direction. This torque impulse would impart an angular pitch rate less than 1 deg/sec to the upper stage during separation. If the J-2 engine thrust vectors can be gimbaled to within a radius of 0.3 ft of the vehicle center of gravity at shutdown, the resulting cutoff torque impulse could be cut down to 26,730 lb ft-sec (corresponding to a 99.7 percent probability level). This latter torque impulse would impart an angular pitch rate less than 0.2 deg/sec to a space stage.

[redacted]

<u>Thrust Vector Control During Space-Stage Powered Flight</u>. Examination and reduction of data formulated in simulated trajectories for the space stage, initiating prior to an Earth orbit and providing powered flight for a lunar mission, have indicated small requirements for pitch and yaw thrust vector control. Providing the correct orientation is established, an engine startup (by the attitude control system) 0.1 to 0.2 deg gimbal angle is adequate. The yaw requirements inherently will be less than pitch requirements since a programmed vehicle pitch attitude is required, and the only yaw requirement is that a constant yaw attitude be maintained.

The vehicle attitude during an Earth escape-phase operation does not greatly effect the mission performance. A vehicle orientation of approximately ±1.0 deg from the nominal (optimum) case throughout the simulated flight resulted in a payload reduction of less than approximately 0.3 percent. Thus high response rates for thrust vector control do not appear to be necessary from a flight control standpoint.

<u>Results</u>. These analyses indicate that thrust vector corrective torque requirements for a space stage will result primarily from engine and vehicle thrust vector misalignment. A gimbal angle of 1 to 2 deg should be adequate, together with an auxiliary roll control system if a single main engine is employed. Engine gimbal requirements for a landing stage must be analyzed with a stage design layout so the vehicle dynamics can be considered; the size, configuration, and center-of-gravity location are prime factors. For a lunar or planetary takeoff stage, atmospheric

winds must also be considered However for lunar takeoff and landing, where little atmosphere exists, review of current Earth booster vehicles (5 to 8 deg with margin of safety) indicates that this magnitude should be feasible for a lunar engine system from both a trajectory control and engine design standpoint

Lateral and Axial Acceleration Loads

A review of possible space engine design loads has been conducted to indicate engine system requirements. The loading conditions may be divided into two categories: (1) those occurring during induced booster operation, and (2) those occurring during space stage operation. Axial loading results from the vehicle acceleration necessary to achieve the velocity required for the mission. Lateral loading occurs during boost phase from wind loading or gimbaling of the boost stage engines to provide a vehicle-corrective torque. During space stage operation, lateral loading results primarily from engine gimbaling or possible alternate thrust vector control systems

Axial Loads The axial loads which could result during boost phase operation are tabulated in Table 3-9 As presented, a two-stage Saturn system could result in thrust-to-weight ratios up to 10 for the Earth orbit establishment mission This high value results because the low propellant weight booster design requires a high second-stage mass ratio and thus the high burnout acceleration The Nova vehicles, based on a more optimum staging arrangement are shown to achieve somewhat lower loading conditions

TABLE 3-9

BOOST PHASE AXIAL LOADS

	Axial Load, g
Saturn	
Two-stage to orbit	9 to 10
Three-stage to orbit	4 to 5
Nova	
Two-stage to orbit	5 to 6.5

The space stage operation results in relatively low actual loads. Figure 3-54 presents the burnout thrust-to-weight ratio which will result for a space stage. Because most space stages will be designed with initial thrust-to-weight ratios between 0.5 and 1.5, and for stage velocity increments between 10,000 and 15,000 fps, axial thrust-to-weight ratios less than 4 can be expected.

Lateral Loads. Lateral loads on the space engine system will be induced during boost phase and space-powered operational phases. Review of the lateral loads which could be induced for a space-stage engine atop the Saturn vehicle indicate a maximum lateral load of 2.7 g resulting from maximum engine gimbaling at boost phase propellant burnout (This results in the maximum angular acceleration of the vehicle). However, this condition is not realistic. Reviewing that maximum gimbal requirements occur at the time of maximum dynamic pressure in the trajectory, a lateral acceleration of 0.25 for the space engine would result.

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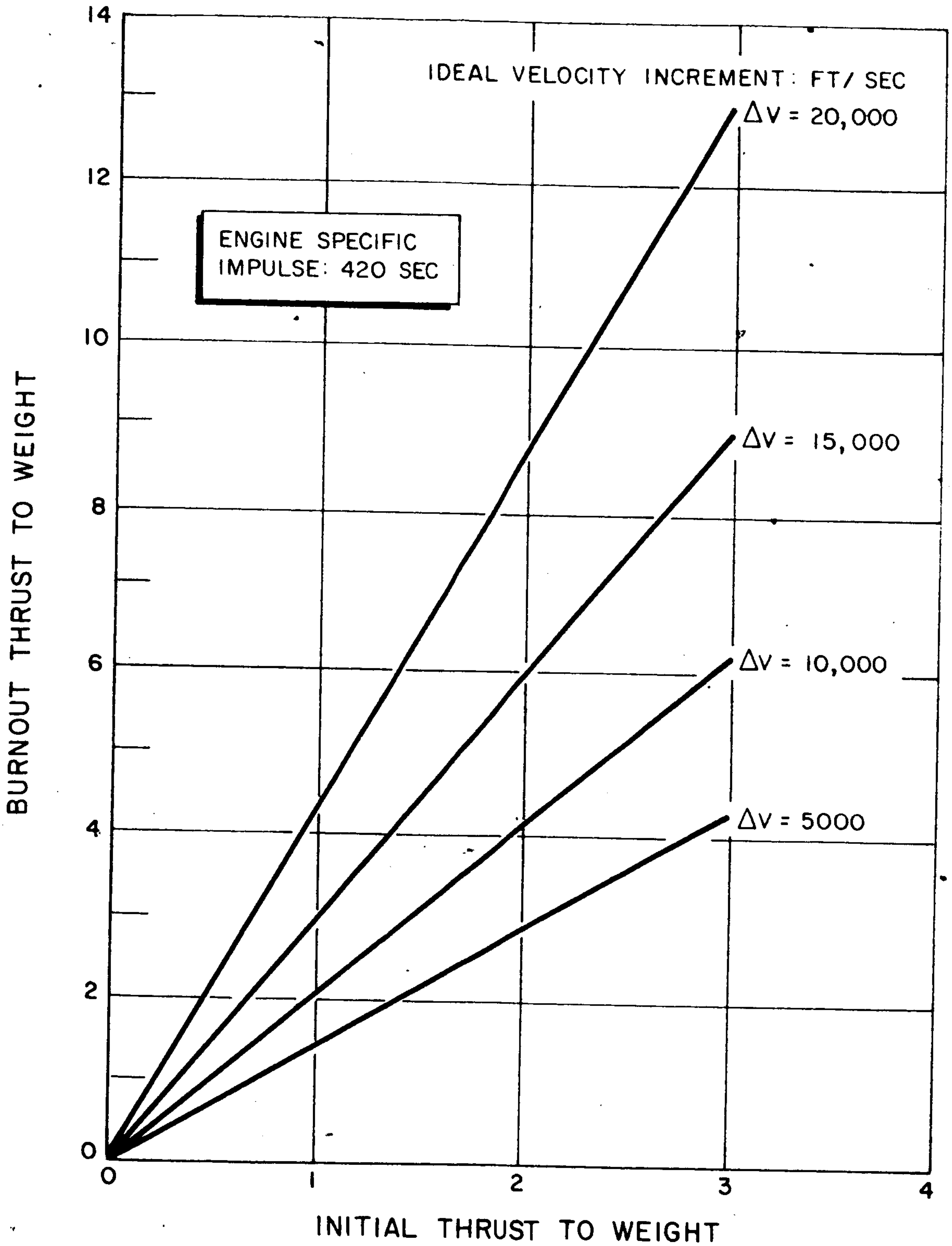

Figure 3-54. Burnout Thurst-to-Weight vs Initial Thrust-to-Weight

[redacted]

During space stage operation, the low thrust-to-weight ratios involved and the small requirement for a gimbal angle for thrust vector control indicate that very low values of lateral acceleration would result in the engine system. For a space vehicle design having a thrust-to-weight ratio of approximately 1, in an Earth orbit escape trajectory, considering a possible engine gimbaling of 3 deg, a lateral loading of less than 0.2 g resulted.

Conclusions. Reviewing the axial loads on a space engine during boost, together with the design specifications for current booster stage engines in Table 3-10, indicates that the space engine stage should be designed to withstand a maximum axial load during boost phase of 8 g. During space stage operation a design for axial loading during booster phase will be adequate.

TABLE 3-10

CURRENT ENGINE DESIGN LOADS

H-1	4 g with 1 g lateral; up to 8 g with 0.5 g lateral
F-1	9 g with 1 g lateral; up to 15 g with 0.5 g lateral
J-2	10 g with 1 g lateral; (4 g any direction ground handling)

Lateral loads of 0.5 g should be adequate for booster or space-stage conditions. Handling loads would result in higher values for the lateral load specification; a handling specification identical to the J-2 criteria, 4 g in any direction for ground handling, appears adequate for a space engine design.

[redacted]

MARS ORBIT MISSION

PHASE 1 MISSION/VEHICLE SELECTION

During the first phase of the NASA contract, several mission/vehicle combinations were considered. For these vehicle/mission combinations general propulsion system requirements were described for performing the various maneuvers of the missions. After review of the first phase, a Mars orbit establishment mission was selected as one of the three missions for study in the second phase. The vehicle selected was the Nova H-6. This vehicle is capable of placing 354,000 lb of payload in a 300 n mi Earth circular orbit. Assuming this payload to be the initial gross weight of a space vehicle, combinations of maneuvers and propulsion systems have been investigated for establishing a Mars orbit.

MISSION MANEUVER ALTERNATIVES

Figure 3-54A shows various combinations of maneuvers available for performing the Mars orbit establishment mission. Wherever various alternatives were available, recommendations based upon analysis in Phase 2 have been shown by shaded areas. A brief summary of the recommended maneuvers follows.

Earth Departure

A departure from an Earth orbit has been selected over direct departure from the Earth's surface. The advantage of this maneuver (discussed in section III of Phase 1) is a relaxation of the stringent spatial alignment

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ESTABLISHMENT

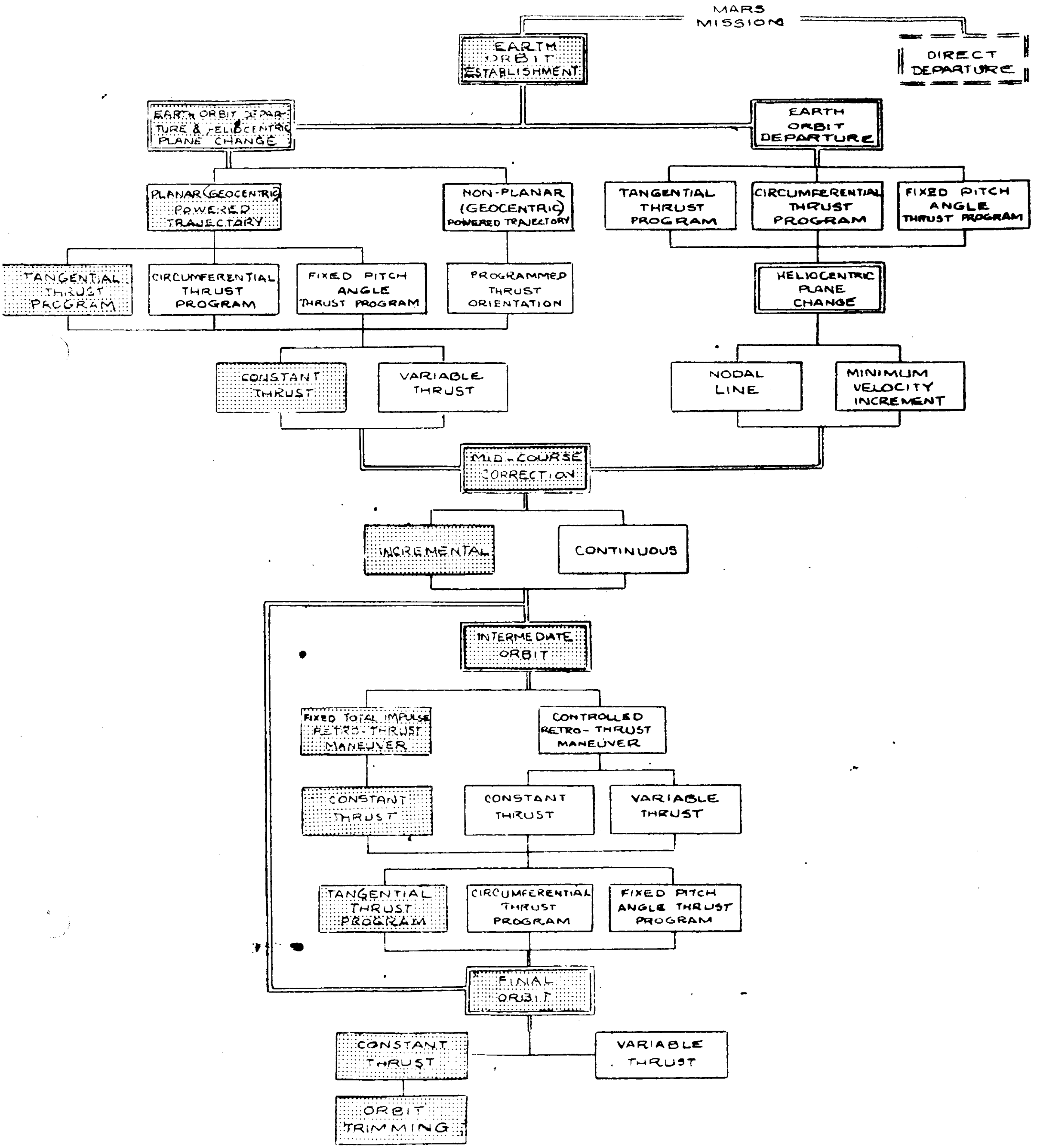

Figure 3-54A. Maneuver Combination for Mars Circular Orbit Establishment constraint between the Earth launch position and Mars. The increased degree of freedom obtained when the vehicle starts an interplanetary transfer from an orbit reduces the propulsion requirements and thus permits an expansion of the launch period interval. Leaving an Earth orbit, two options exist for performing the plane change associated with the majority of interplanetary transfers.

The vehicle may depart from a 28-1/2 deg Earth orbital plane as established by an Atlantic Missile Range launch without changing geocentric planes during the departure maneuver. Excluding extremely special cases, a heliocentric plane change would then be necessary some time after launch before intercepting Mars. The space vehicle could depart the Earth orbit and change heliocentric planes during the same propulsion phase by using a component of thrust normal to the geocentric plane.

A better technique for performing a Mars mission would be to depart the geocentric orbit and change heliocentric planes (as described in the first phase of the study) with a planar propulsion maneuver. A geocentric orbit can be established by the Earth booster vehicle such that the geocentric escape propulsion phase can be accomplished with a planar (geocentric) powered trajectory. The thrust vector is always in the plane defined by the radius and velocity vectors during a planar propulsion maneuver. It is recognized that for this method of escape from the Earth, the geocentric orbit must be controlled and a geocentric orbital plane inclination established which is suitable for this recommended escape maneuver. Establishing a geocentric orbit which is inclined at some angle other than 28-1/2 deg to the equatorial-plane will place a greater requirement upon the boost phase of the Nova H-6. In considering the Mars mission starting from an Earth orbit, the techniques of establishing the correct geocentric inclination have been omitted.

Escape from the Earth orbit with a geocentric planar trajectory can be performed by a variety of thrust programs. A tangential thrust program (thrust parallel to velocity) is recommended for this Earth orbit departure phase. Near optimum performance is obtained using this thrust program. Additionally, the thrust level of the propulsion system has been assumed as constant. In Phase I of the study, a 150,000-lb-thrust propulsion system was recommended for powering the first stage of a space vehicle of 354,000 lb gross weight.

Mid-Course Correction

After the Earth escape heliocentric plane change propulsion maneuver, the next propulsion requirement (neglecting attitude control) is for midcourse corrections. The cutoff impulse deviations of the constant thrust engine of the first stage (the Earth departure stage) add to the errors in position and velocity measurements by the guidance and control equipment, and introduce errors in the resultant heliocentric trajectory at burnout. To compensate and correct for these errors at burnout, some midcourse propulsion is necessary. To modify the trajectory sufficient measurements must be acquired to ascertain the magnitude of the deviation from the standard trajectory.

Incremental midcourse correction (applied in as few as two increments) is recommended to permit sufficient time to elapse between corrections to obtain trajectory errors. One correction is applied shortly after launch; a second, some time shortly prior to entering the Mars sphere of influence. The alternative to incremental corrections is a continuously applied correction maneuver. A continuous midcourse correction system would require an extremely low thrust engine such as an ion propulsion system, with its large power supply weight; or a cold gas system of low performance which would require a relatively large expenditure of propellants. Therefore, an incremental midcourse correction maneuver offers greater advantages in payloads (unless a power supply adequate for an ion system is an intrinsic part of the payload).

Mars Orbit Establishment

Arriving in the vicinity of Mars, the vehicle has hyperbolic velocity with respect to a Martian coordinate system. Unless some propulsion is applied to reduce the velocity of the vehicle, it will follow a hyperbolic path past Mars and continue into space. A retrothrust system is required to reduce this velocity. The retrothrust propulsion system can change the hyperbolic approach trajectory directly into the selected Mars orbit.

For direct establishment of the final orbit, the entry corridor influences the propulsion requirements. For ideal entry corridor conditions, i.e., assuming that the asymptotic approach distance has been matched to the hyperbolic approach velocity by the midcourse correction propulsions phases; and that the exact altitude for initiating retrothrust has been determined, the second stage of the space vehicle can enter the final orbit with a minimum expenditure of propellant. If these conditions are not satisfied, the propellant requirements increase for changing the trajectory from the hyperbola to a final orbit.

Since the hyperbolic approach velocity varies with the launch date, it is recommended that an intermediate orbit be established which later can be modified into the final orbit. Establishing this intermediate orbit allows greater flexibility in the retrothrust program for the mission. Since an intermediate orbit is being established, a constant thrust propulsion system would satisfy the requirements of this maneuver. The deviations in engine operation characteristics would only alter the intermediate orbit. Again, a tangential thrust orientation program of thrust opposing velocity at all times would be employed.

After the second stage retrothrust system has effected capture into an intermediate orbit, sufficient time exists for guidance and control systems to measure precisely the orbit and to determine the necessary corrections to establish the final orbit. From this intermediate orbit, the vehicle could possibly compute its own correction, or relay information to the Earth for the vast computer facilities based on the Earth to determine the intermediate orbital elements. The necessary information could be sent back to the satellite for applying the propulsion phases for establishing the final orbit.

For establishing the final orbit a variable thrust engine may be used to minimize the effect of cutoff deviations and thus more precisely control the final orbit. An alternative and the recommended method would use one of the second stage constant thrust propulsion engines (assuming that it could be restarted) for establishing the final orbit, and then some much smaller engine for orbit trimming.

Parallel staging could possibly be used with the latter concept to effect a gain in payload. This staging technique was not included in the analysis performed in Phase 2.

If the liquid oxygen/liquid hydrogen propulsion system used for the intermediate orbit establishment could not be restarted, it would be necessary to use another stage engine system for establishing the final orbit.

The broad categories for establishing a Mars orbit have been defined in a preceding paragraph and the recommendations for performing these maneuvers have been cited. The recommendations are as shown in Fig. 3-54A by shaded areas.

EARTH ORBIT DEPARTURE

The 1964 period of launch dates was chosen for study in Phase 2. From Phase 1 of the study, during the 1960 launch period for a Mars space vehicle, a 200 day transfer time was shown to have approximately the minimum energy requirements (Fig. 2-79 , Section III). For the 1964 launch period analysis, variations of ±20 days in trip times about the nominal 200 day transfer time were assumed.

Figure 3-55 presents the Earth hyperbolic excess velocity of the Mars bound vehicle as a function of the launch date during the 1964 period. Curves are plotted for the nominal 200 day transfer and for 180 and 220 day transfers. Similar graphs were shown in Section III for a launch during the 1960 period. For comparison purposes, similar curves have been generated for the launch period which occurs in 1966 to 1967 range of dates. These are represented in Fig. 3-56 . From Fig. 3-55 for the 1964 launch period, a 220 day transfer time requires less energy for Earth departure than for the 200 day transfer time. However, for the 1966 to 1967 group of launch dates, the 200 day transfer time

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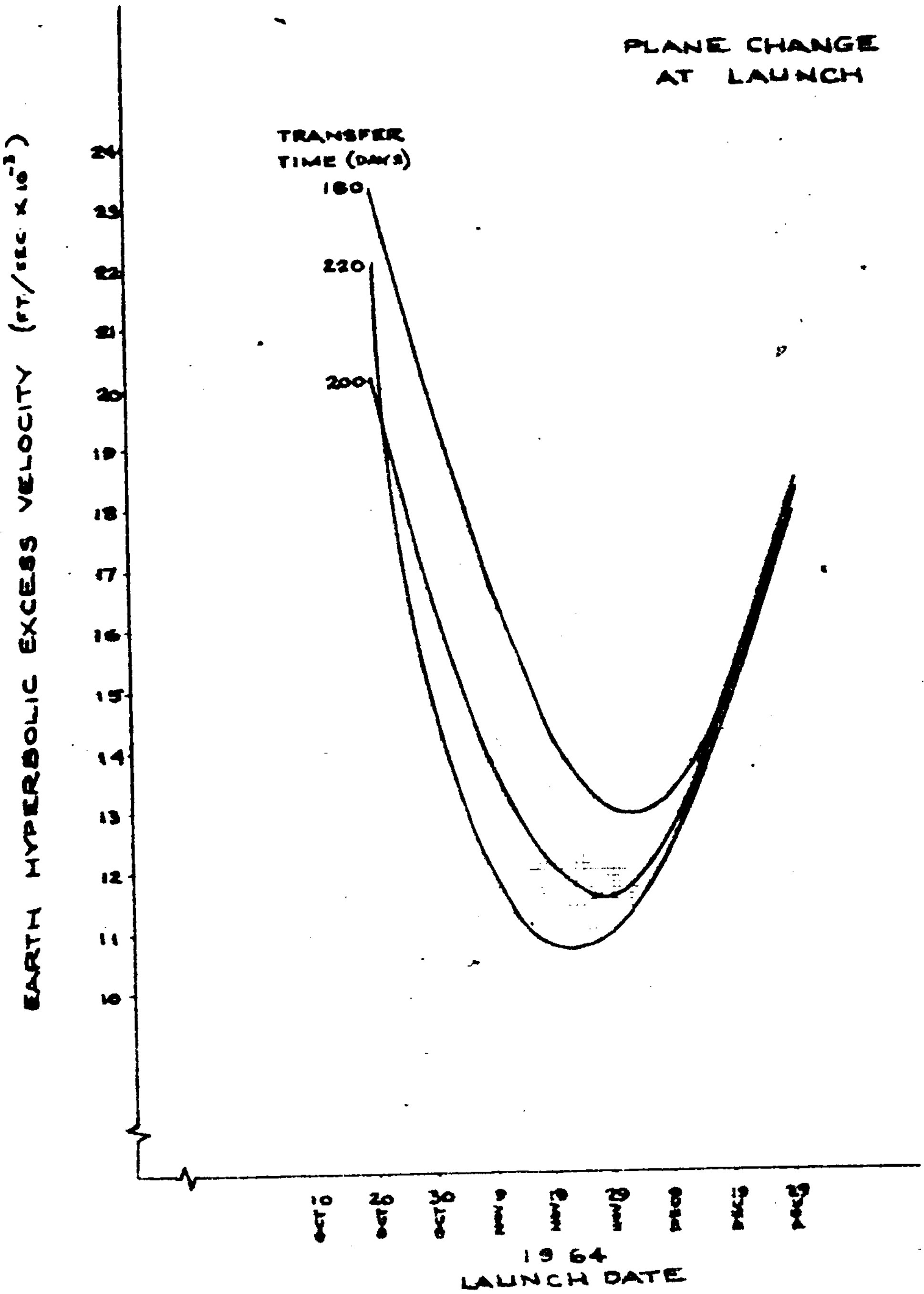

Figure 3-55. Earth-Mars Planetary Mission

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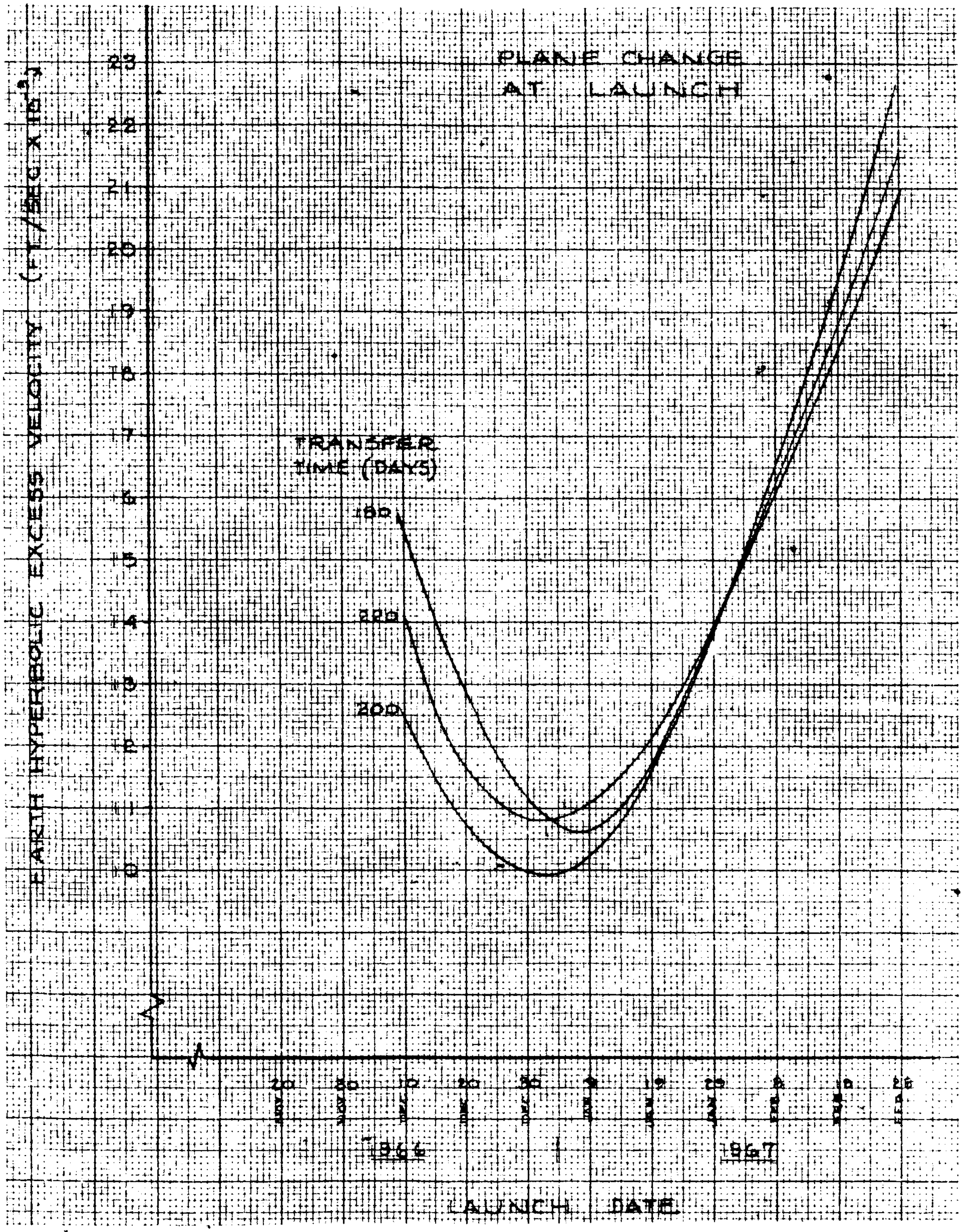

Figure 3-56. Earth-Mars Planetary Mission

requires less energy than the 220 day transfer time. From reference sources,* data are available to show how the transfer times, corresponding to minimum energy expenditure, vary on either side of a 200 day transfer time according to the year of launch (Fig. 3-57). Therefore, the 200 day trip has been selected as the nominal case to permit a vehicle configuration to be used during each launch period without extreme performance penalties.

Corresponding to the graphs for the Earth departure phase, Fig. 3-58 and 3-59 present the hyperbolic arrival velocity at the planet Mars. These curves are plotted as functions of the Earth launch date. The velocities represent the vehicles hyperbolic approach at Mars after the elapse of the indicated transfer time. The hyperbolic arrival velocities at Mars and the hyperbolic departure velocities at Earth during the 1964 optimum period do not reach minimums simultaneously. Trade-off studies were conducted to determine the launch date within the period to give maximum over-all vehicle performance.

In the first phase of the study, nomographs were presented as a tool for the analysis of vehicles performing interplanetary missions. Nomographs constructed for Earth, Mars and Venus, had specific impulse ranges between 200 and 800 sec. In the second phase, for designing a specific vehicle a more detailed analysis of the influence of I_s was required for better definition of payloads. Therefore, for Mars and Earth, nomographs have been constructed expanding the region of specific impulses between 300 and 440 sec. Any vehicle launched in the immediate future (the years considered in the study: 1964 and 1967) probably will have propellant combinations giving specific impulses somewhere in this range.

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*1. Breakwell, J. V., et al, "Hyperbolic Excess Speeds for Trips to Mars, Astronautical Sciences Review, April/June, 1961.

2. Breakwell, J. V., et al, "Researchs in Interplanetary Transfer, Journal of The American Rocket Society, 954-59.

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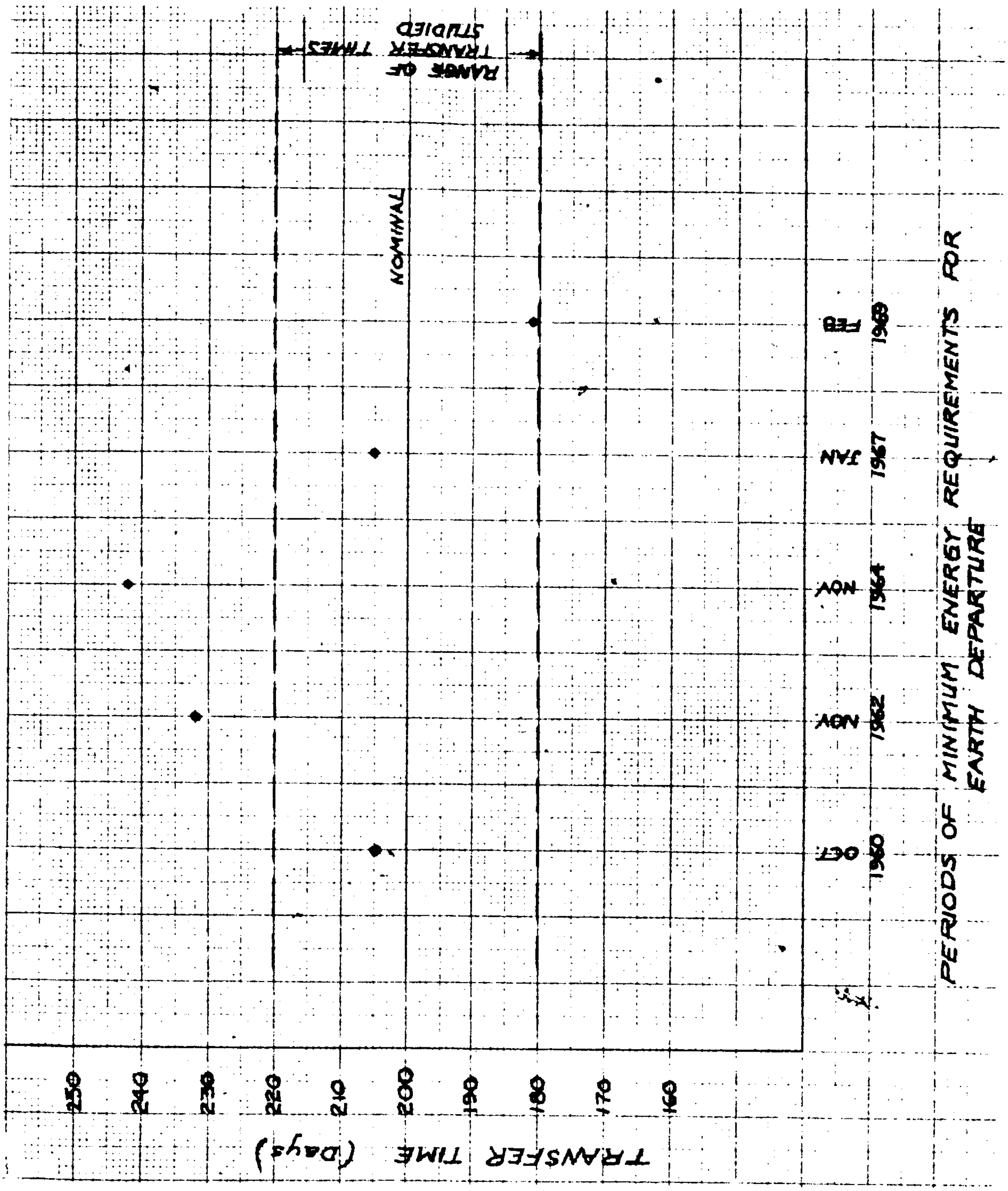

Figure 3-57. Earth-Mars Transfer

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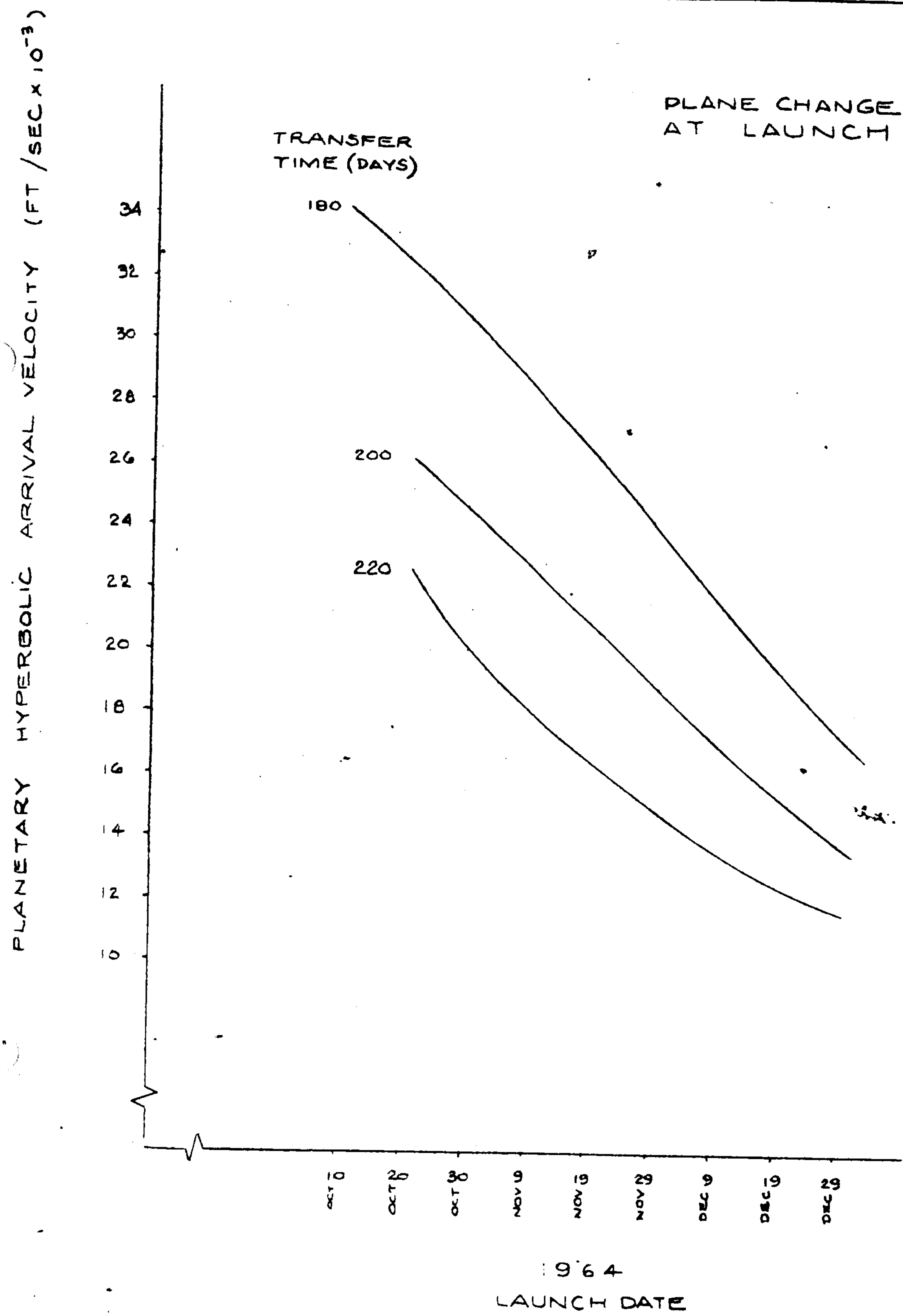

Figure 3-58. Earth-Mars Planetary Mission

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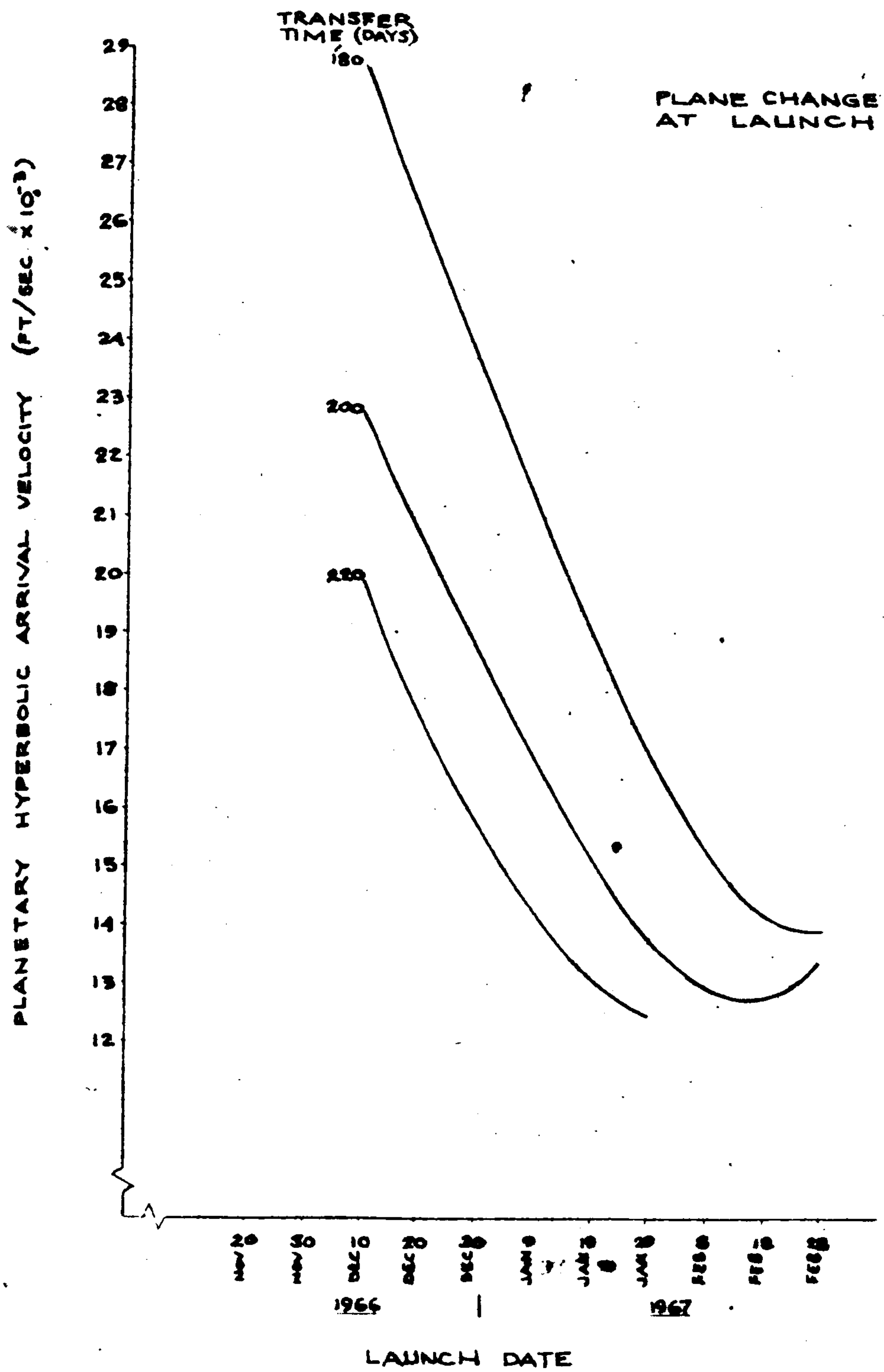

Figure 3-59. Earth-Mars Planetary Mission

A second stage, or the Mars orbit establishment stage, design for storable propellants will have an I_s in the 300 category; a high-energy liquid oxygen/liquid hydrogen system employed in the second stage will have a specific impulse in the 420 range. Thus, these new nomographs should cover any propellant combination anticipated for a Mars orbit establishment mission over the next decade. These new nomographs are shown in Fig. 3-60 and 3-61.

Throughout the analysis of this mission a first and second stage propellant fraction (λ_p) of 0.915 will be used for this nominal vehicle of 354,000 lb gross weight. Later, vehicle performance with the assumed λ_p will be compared with results obtained using a preliminary design stage weight, as determined in the second phase of this study. For a study of relativistic vehicle/propulsion system performance it is valid to use an assumed propellant fraction.

For the recommended Earth departure maneuver using the planar powered trajectory, the initial gross weight of the second stage is shown in Fig. 3-62 as a function of the launch date and in Fig. 3-63 as a function of Mars hyperbolic arrival velocity. The second stage gross weight is shown for a first stage specific impulse of 420 sec (characteristic of liquid oxygen/liquid hydrogen systems). For comparison, the gross weight of the second stage has been shown for a first stage delivering a specific impulse of 400 sec. A noticeable decrease in the second stage gross weight is shown for the loss of 20 sec specific impulse in the first stage.

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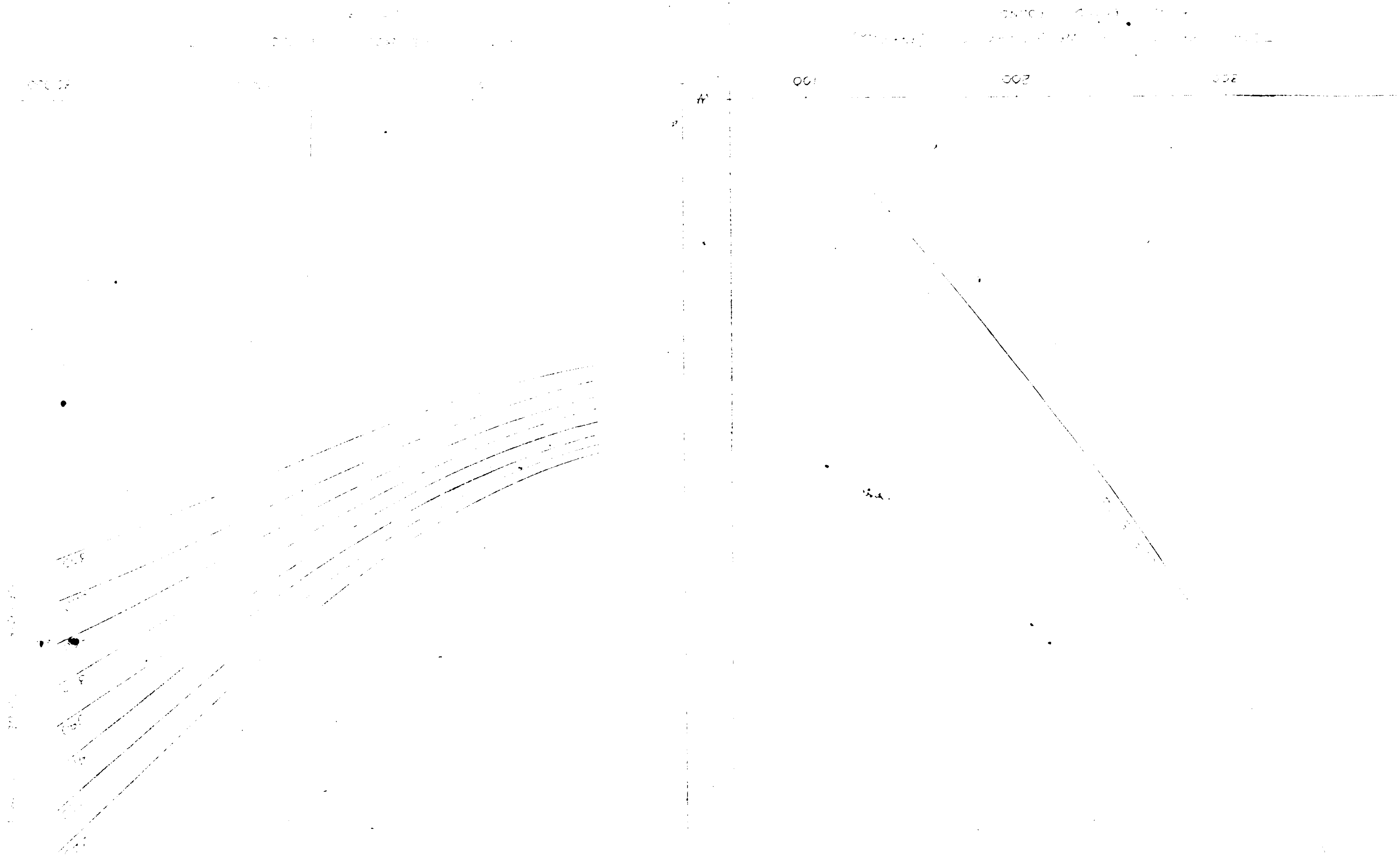

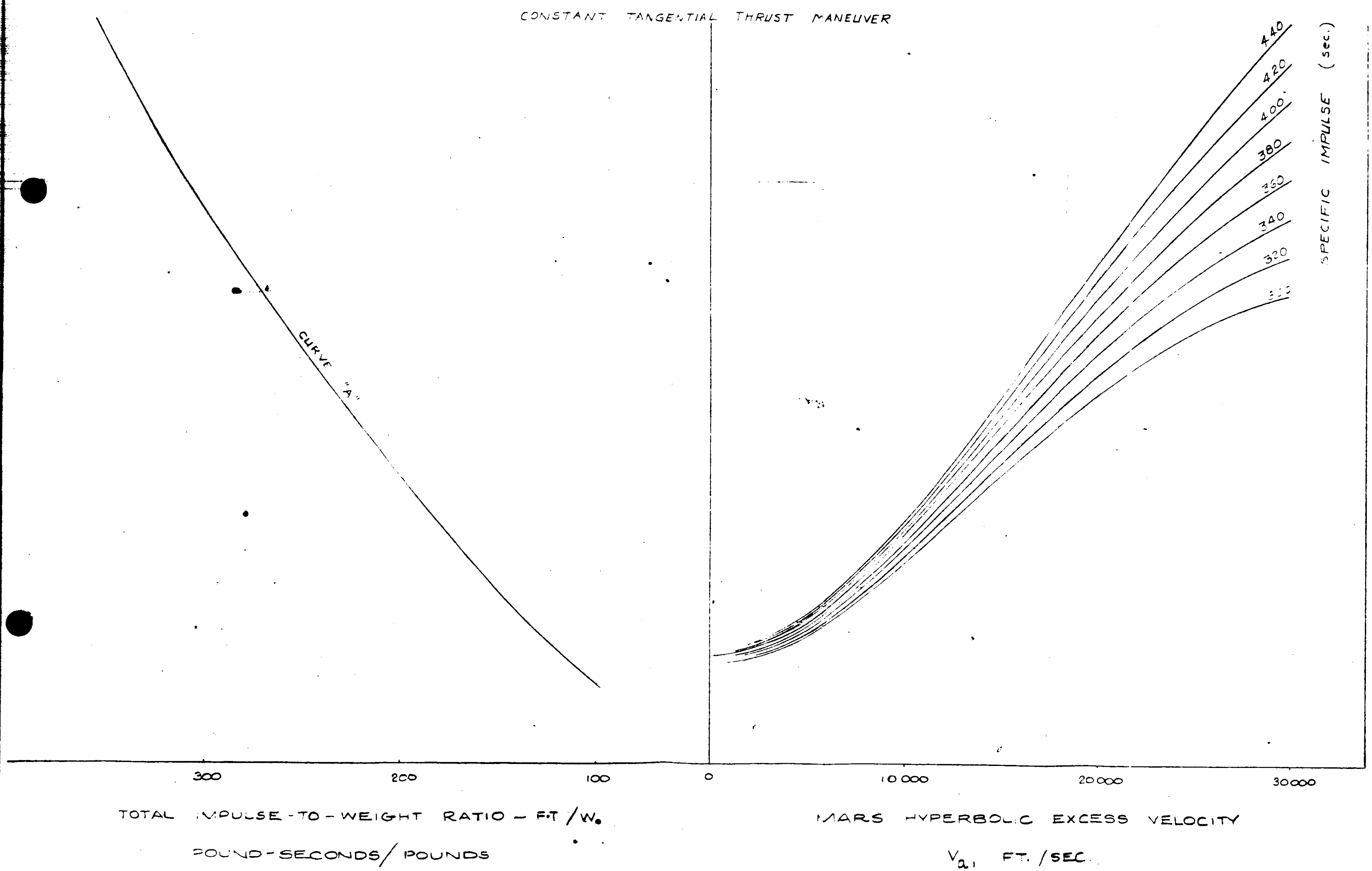

Figure 3-61. Mars Hyperbolic Excess Velocity Nomograph

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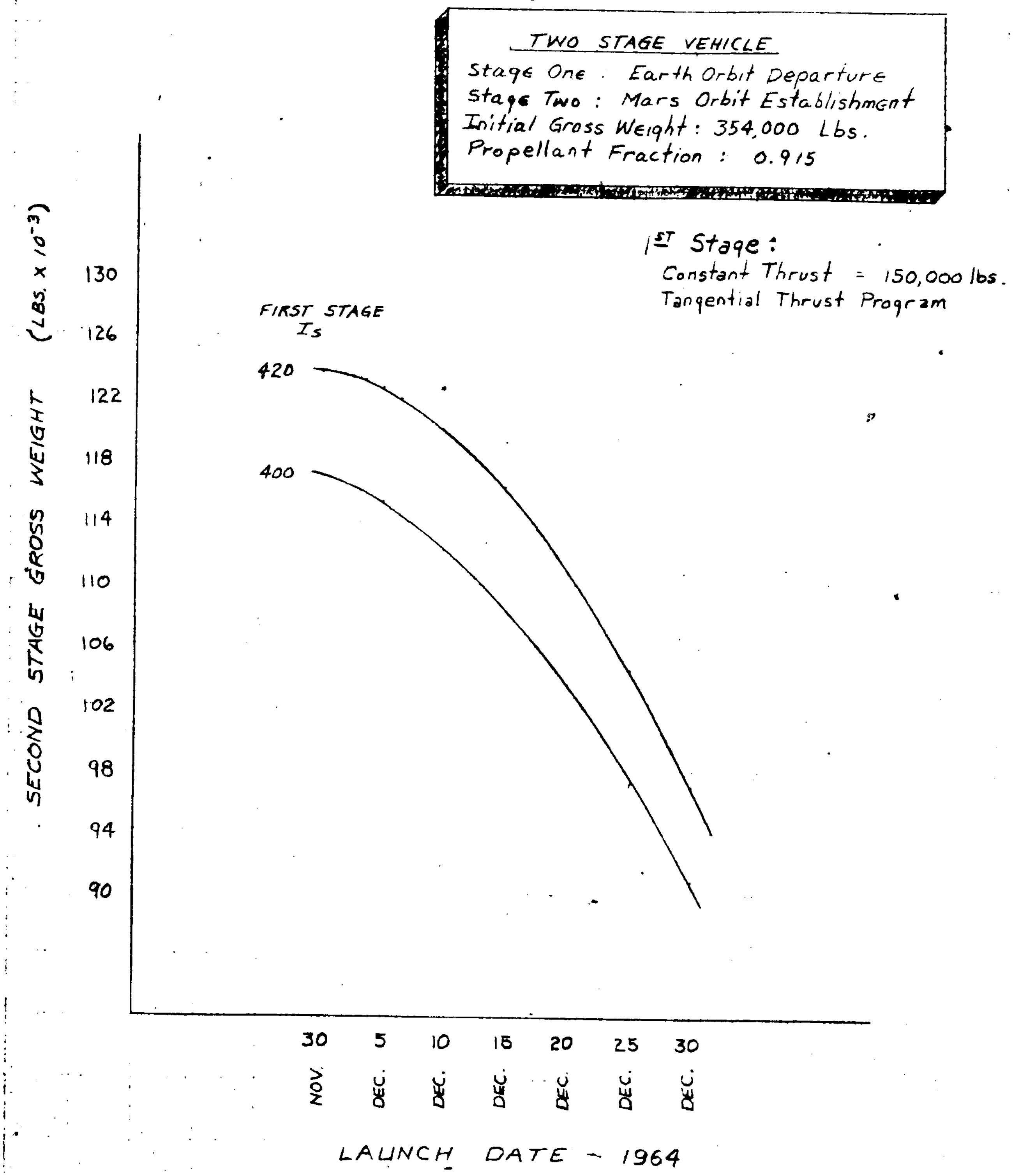

Figure 3-62. 200 Day Mars Transfer

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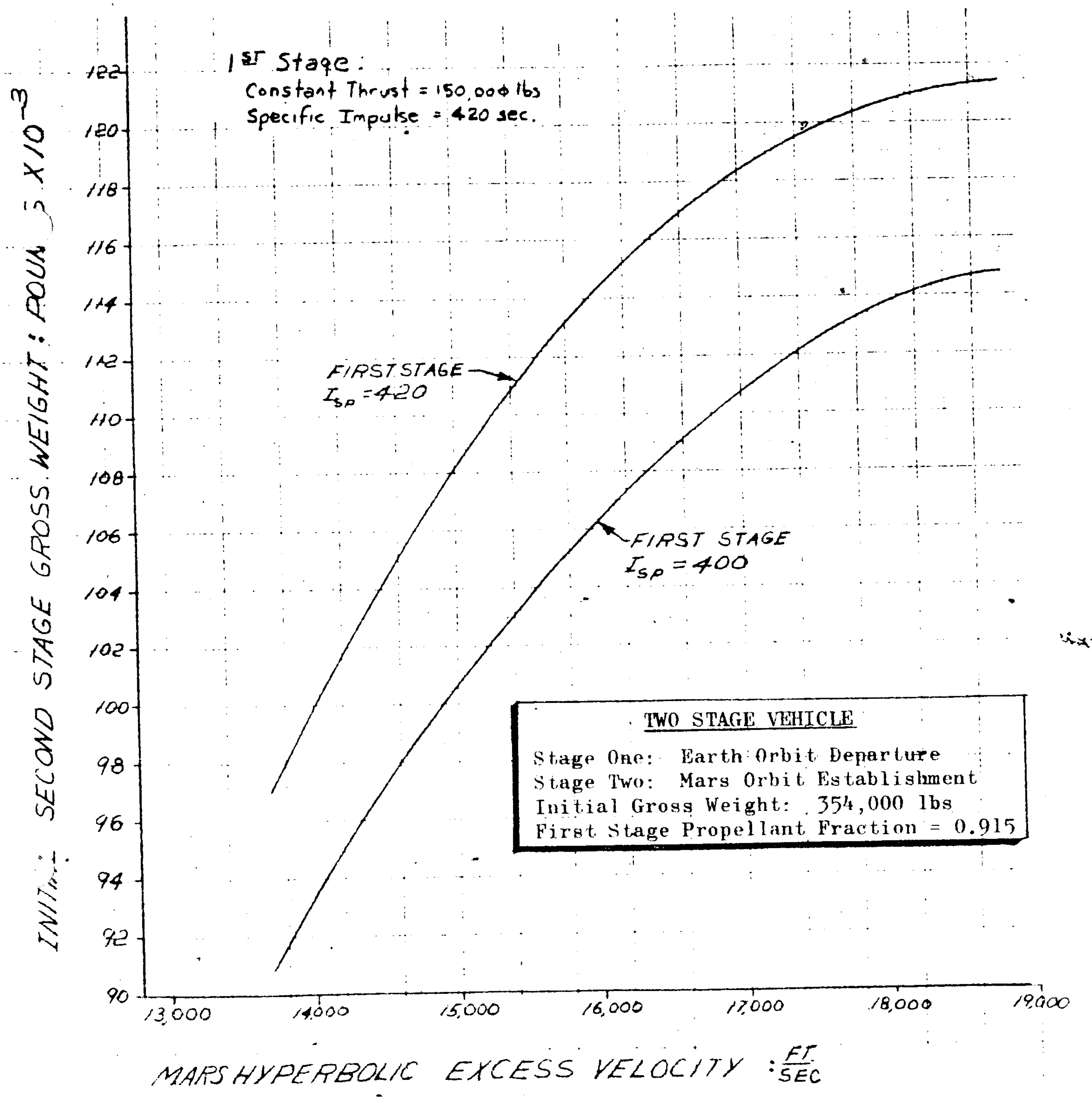

Figure 3-63. 200 Day Mars Transfer (1964 Launch)

MIDCOURSE CORRECTION

Space Transfer Propulsion Phase

The next phase of propulsion required, after the vehicle has escaped the Earth orbit and is travelling in its heliocentric transfer trajectory is, midcourse correction. The time delay between launch and the application of the midcourse correction velocity increment influences the size of the velocity increment. There is a trade-off however, between the delay time when the velocity increment is applied and the merit of the results of the correction. If the correction increment is applied immediately after launch the velocity increment will be small, depending, of course, on the size of the burnout error. If the correction is applied later in the trajectory, when more information is available to more accurately ascertain the magnitude of the difference between the actual trajectory and in intercept trajectory, a larger velocity increment is necessary.

Errors in the magnitude of the injection heliocentric velocity vector and errors in the heliocentric elevation angle at injection have been examined for their influence upon the size of a midcourse correction applied shortly after departing the Earth orbit. This midcourse correction is the first of two recommended midcourse corrections. The second correction is applied shortly prior to entering the Mars sphere of influence to correct for any perturbing effects upon the vehicle during its coast trajectory to the vicinity of Mars. In addition, a second correction nullifies any errors in applying the first midcourse correction.

For an examination of midcourse corrections, a simplifying assumption will be made. One parameter at a time will be evaluated for effects of errors. At the first midcourse correction, Fig. 3-64 relates the required velocity increment as a function of the percent error in the magnitude of Earth hyperbolic excess velocity. To show the dependency between delay time of this midcourse correction, parametric variations (20, 30, 40, and 50 days) in the delay time have been analyzed. For errors less than 0.5 percent in the hyperbolic excess velocity magnitude and a delay time of up to 50 days, the maximum correction velocity increment is approximately 100 ft/sec. For the nominal vehicle, the 0.5 percent deviation represents a variation of about 66,800 lb-sec in total impulse for a 30 November 1964 Earth orbit departure date. After 20 days, a sufficient time should have elapsed for measurements to be made to evaluate the error in the heliocentric trajectory. From Fig. 3-64, for a 20 day delay time between launch and midcourse correction, an error of about 0.8 percent in the hyperbolic excess velocity magnitude could be tolerated for the 100 ft per sec correction velocity increment capability. Again, this velocity increment assumes no error existed at burnout in the elevation angle of the velocity vector. Additionally, it assumes no error existed in the inclination of the resultant heliocentric transfer plane from the intended heliocentric transfer plane.

Figure 3-65 shows the midcourse correction velocity increments required for errors in the injection heliocentric elevation angle. Again, curves are plotted for 20, 30, and 40 day delay times to show the effect of obtaining more accurate information vs penalty in larger velocity requirements. Considering the delay time of 20 days and a 1 percent error in the elevation angle, approximately 50 ft per sec ΔV will be required. In actuality, errors will exist at burnout in the heliocentric elevation

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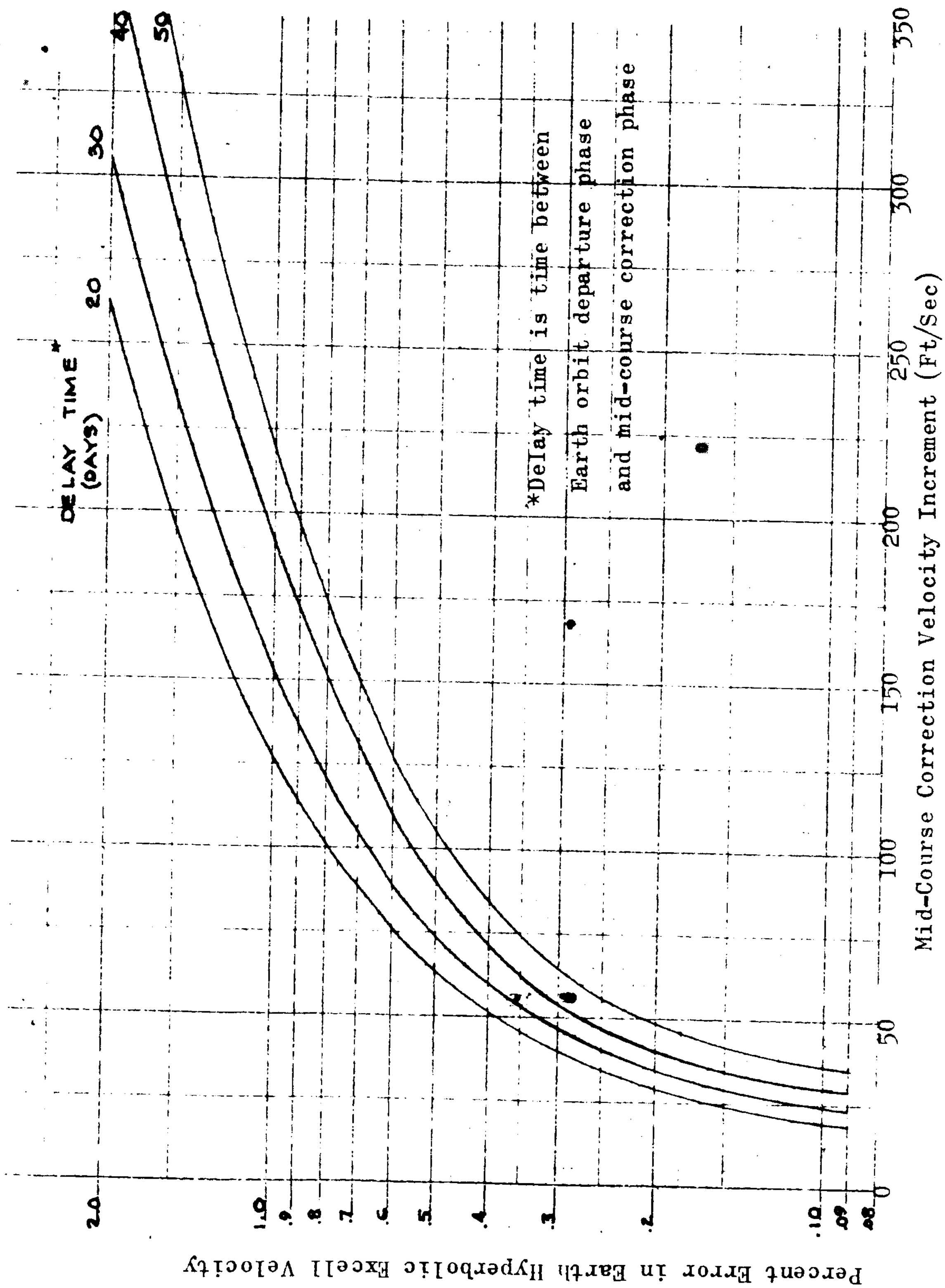

Figure 3-64. 200 Day Mars Transfer (1964 Launch)

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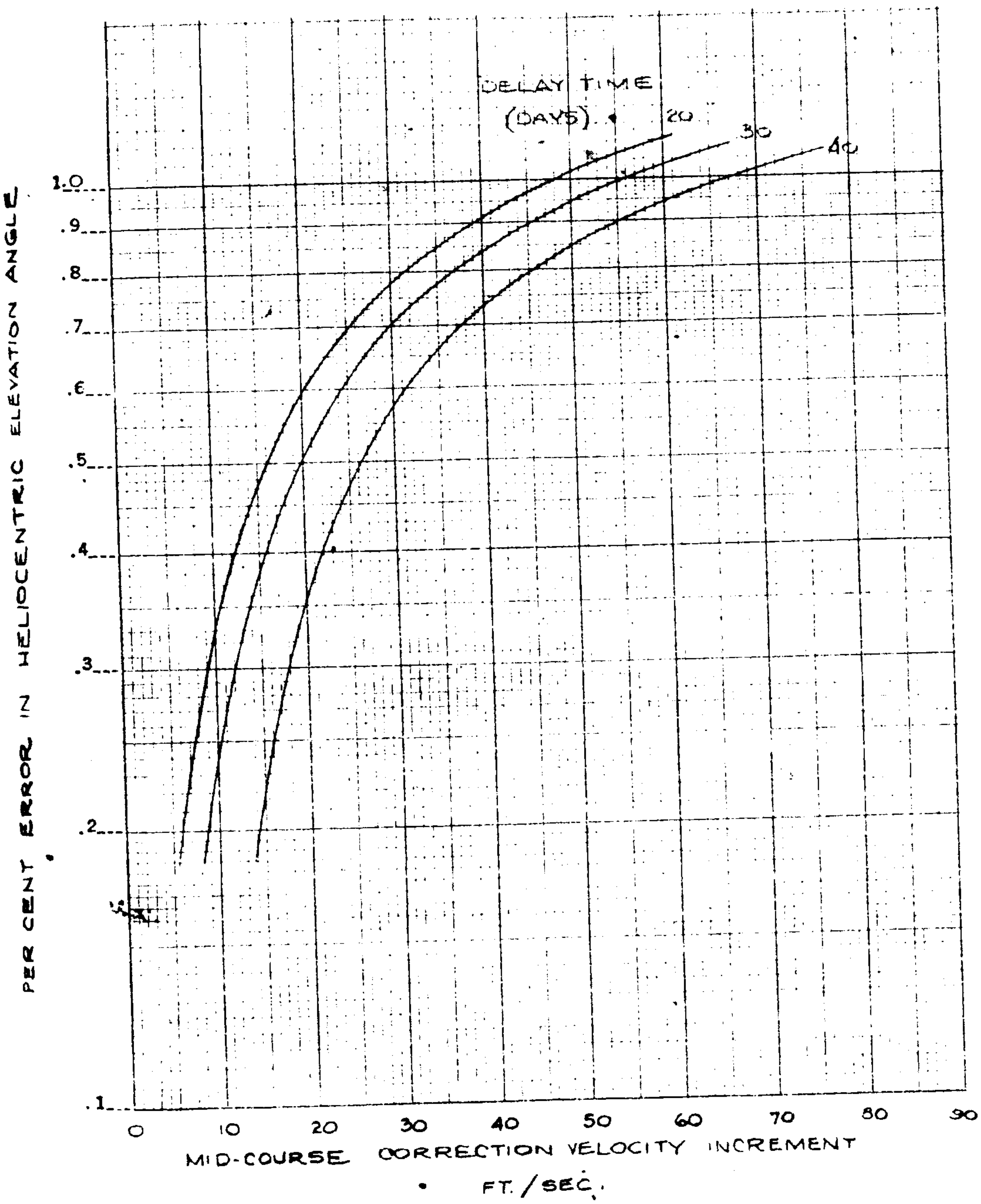

Figure 3-65. 200 Day Mars Transfer (1964 Launch)

angle; in the magnitude of velocity vector; in the angular position of the vehicle; and in the inclination of the orbital plane. Therefore, the first midcourse correction increment will be a resultant of a combination of errors in each of these quantities. For a delay time of 20 days, the midcourse correction can be determined from Earth-based computer facilities. The actual magnitude of the midcourse correction at this first correction should be less than 100 ft/sec, assuming reasonable burnout control of the Earth orbit departure phase. The second midcourse correction will be applied near the vicinity of Mars, if required.

Entry Corridor Correction Phase

To gain some information about the magnitude of the second midcourse correction (entry corridor maneuver), a nominal case has been selected for study. The 354,000-lb space vehicle is launched on 30 November 1964. The burnout velocity of the space vehicle, using conventional guidance equipment can be controlled to ±4 ft/sec*. However, as a safety factor, a velocity burnout error of ±10 ft/sec will be assumed. Figure 3-66 relates the change in burnout velocity of the rocket to the resultant hyperbolic excess velocity. This is an energy derived relation.

For the 30 November 1964 launch date, a hyperbolic excess velocity of approximately 11,600 ft/sec is required. The assumed error of ±10 ft/sec in the burnout velocity gives approximately ±35 ft/sec deviation in the hyperbolic excess velocity.

*Spence, W. N., "On the Adequacy of ICBM Guidance Capability for a Mars Launch," Journal of the American Rocket Society, 1174-60.

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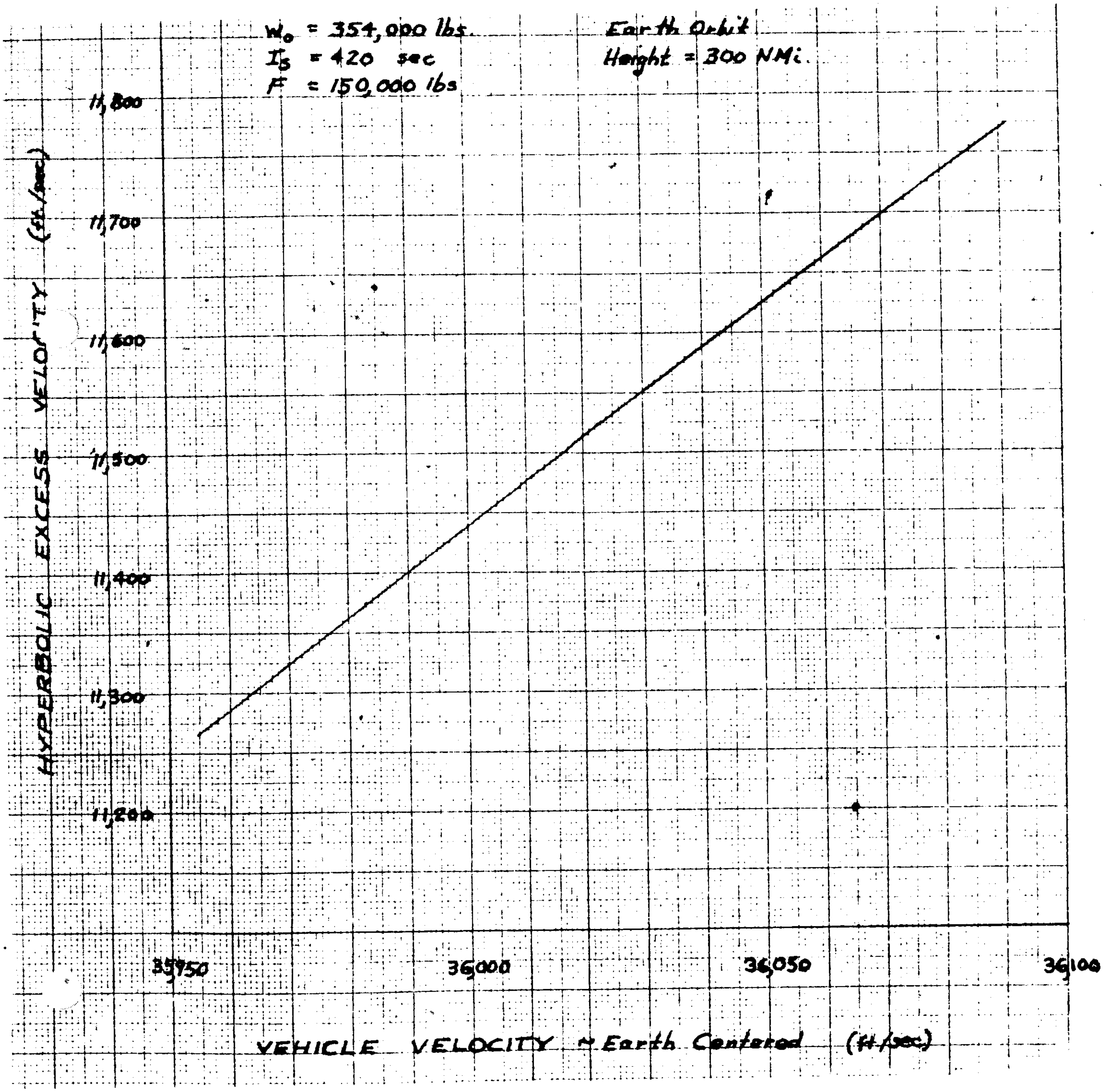

Figure 3-66. Mars Orbit Mission, Earth Departure Phase, 30 November 1964

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For this deviation in the Earth departure conditions, an initial midcourse correction was applied after a 20 day delay time. The first correction reorients the vehicle's heliocentric trajectory to establish a designated asymptotic approach distance at Mars. In applying the first correction maneuver, deviations occur between the required velocity increment and the achieved increment. These deviations necessitate a second correction increment in the vicinity of Mars to reorient the vehicle trajectory to establish the asymptotic approach distance while maintaining the original transfer time.

In the analysis, simplifications have been used to evaluate the magnitude of the second correction increments:

1. The measurements of the vehicle trajectory parameters are assumed correct all times.
2. The radius, velocity, and velocity increment vectors are assumed coplanar.

With these simplifications, the entry corridor correction velocity magnitude is a function of the error in the first correction maneuver and of the remaining time period (preintercept time) between the selected transfer time and the elapsed travel time. Figure 3-67 presents this relationship for corrections applied to re-establish the selected asymptotic approach distance to affect an orbit establishment maneuver. The selected asymptotic approach distance is for a constant retrothrust maneuver into a 300 n mi circular orbit after a 200 day transfer beginning with Earth orbit departure on 30 November 1964.

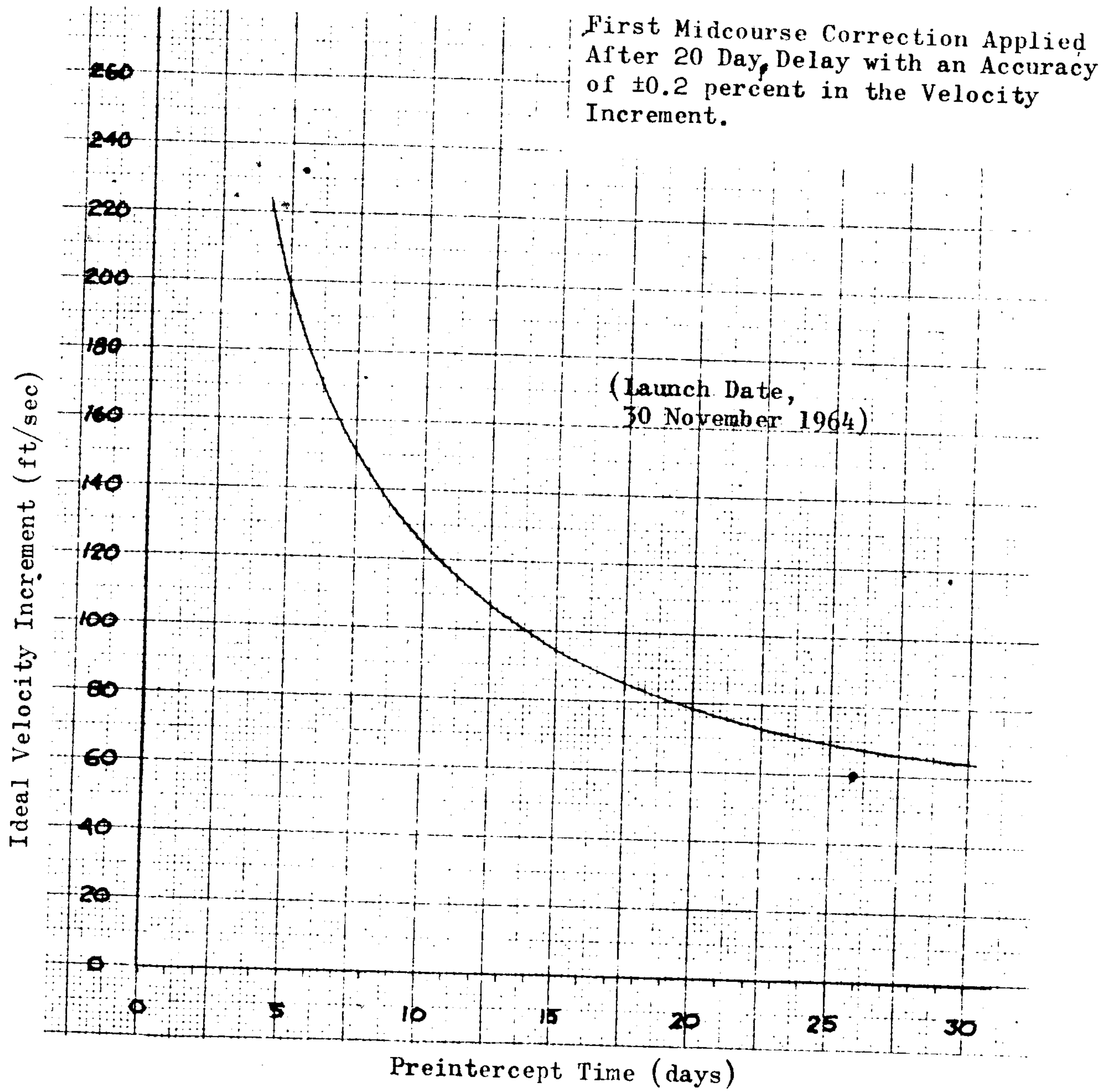

Figure 3-67. Second Midcourse Correction (Entry Corridor Establishment) 200 Day Mars Transfer

MARS ORBIT ESTABLISHMENT

Mars Orbit Selection

One mission/vehicle selection for study in Phase 2 was establishment of a satellite orbit about Mars. A circular orbit of 300 n mi altitude was selected arbitrarily for study of the propulsion system requirement. To select the optimum altitude for an orbit would involve a detailed study of the goals of the mission. This is beyond the scope of the present study. However, the 300 n mi orbital altitude selection was examined for atmospheric decay of the orbit and for photographic capabilities by the orbiting vehicle. These literature review studies indicated the selection of 300 n mi appears very favorable and realistic.

Martian Atmosphere. The density of the Martian atmosphere plays an important role in selecting the orbital height of the space vehicle. The diameter of Mars is less than that of the Earth, and Mars is less massive. Because of the lesser gravitational pull of Mars, the pressure gradient with altitude in the Martian atmosphere is about 2-1/2* times as small as for the Earth. At about 30 km altitude the barometric pressures of Earth and Mars are equal. Below this altitude the Martian atmosphere is less dense and at greater altitudes the atmosphere is more dense.

*Wanders, A. J. M., "The Physical Conditions on the Planet Mars," IX International Astronautical Congress; Amsterdam 1958.

The atmosphere of the Earth is sufficiently rarified at 300 n mi to prevent rapid decay of a satellite orbit. Although the density of a Martian atmosphere will be slightly greater at the 300 n mi altitude, it is felt that the orbit decay will be sufficiently low to provide an extended period for satellite observation of the planet. No concentrated effort was devoted to the Martian atmosphere study, but the literature surveyed has not disproved the undesirability of the selected altitude.

Photographic Capabilities. In the one-way mission to establish a Mars orbit, certain payload operations will be carried out during the orbiting. One such operation involved in a Mars reconaissance mission would be photography of the Martian terrain. The photographic capability from the orbiting space vehicle may greatly influence the selection of an orbiting altitude. A Mars reconnaissance orbit of 300 n mi is examined for feasible observation and camera coverage of the Martian terrian.

A cursory check of photographic capabilities was made based on literature information. Certain fundamentals require review before feasible orbital altitudes can be selected.

Scale Number. This is the number by which a distance on a map or photograph must be multiplied to get the corresponding ground distance and is essentially

$$S_N = \frac{\text{Altitude in feet}}{\text{Focal length in feet}}$$

[redacted]

Figure 3-68 shows the effect of focal length on photograph scale which is shown as n mi per inch of photograph to provide a guide for selection of photograph size. In general, the larger the scale number, the harder it is to see fine detail; on Fig. 3-68 small values of n mi per inch are desirable. Figure 3-68 also shows an actual photograph result taken by camera from a Viking flight which can be considered early state of the art.

Resolution. Another useful and measurable parameter is resolution, a term originally used by astronomers in specifying the ability of a telescope to visually separate double stars. For photographic performance, resolution refers to the ability of a film-lens combination to render barely distinguishable a specified target consisting of black and white lines. When a lens-film combination yields a resolution of 10 lines/mm, it means that a line and a space which together measures 0.1 mm are barely distinguishable. Lines of coarser spacing (i.e., 0.1 mm) are, therefore, better resolved. This single parameter (resolution in lines/mm) should be used with caution for it fails to describe the character of the resolution at all points other than the last, or threshold value; but if used properly it is a very handy measure.

Ground Resolution. This term is often used in discussing performance and is defined as the ground size equivalent to one line at the limit of resolution or

$$\text{Ground resolution (feet)} = \frac{\text{Scale number}}{304.8\ R\ (\text{lines/mm})}$$

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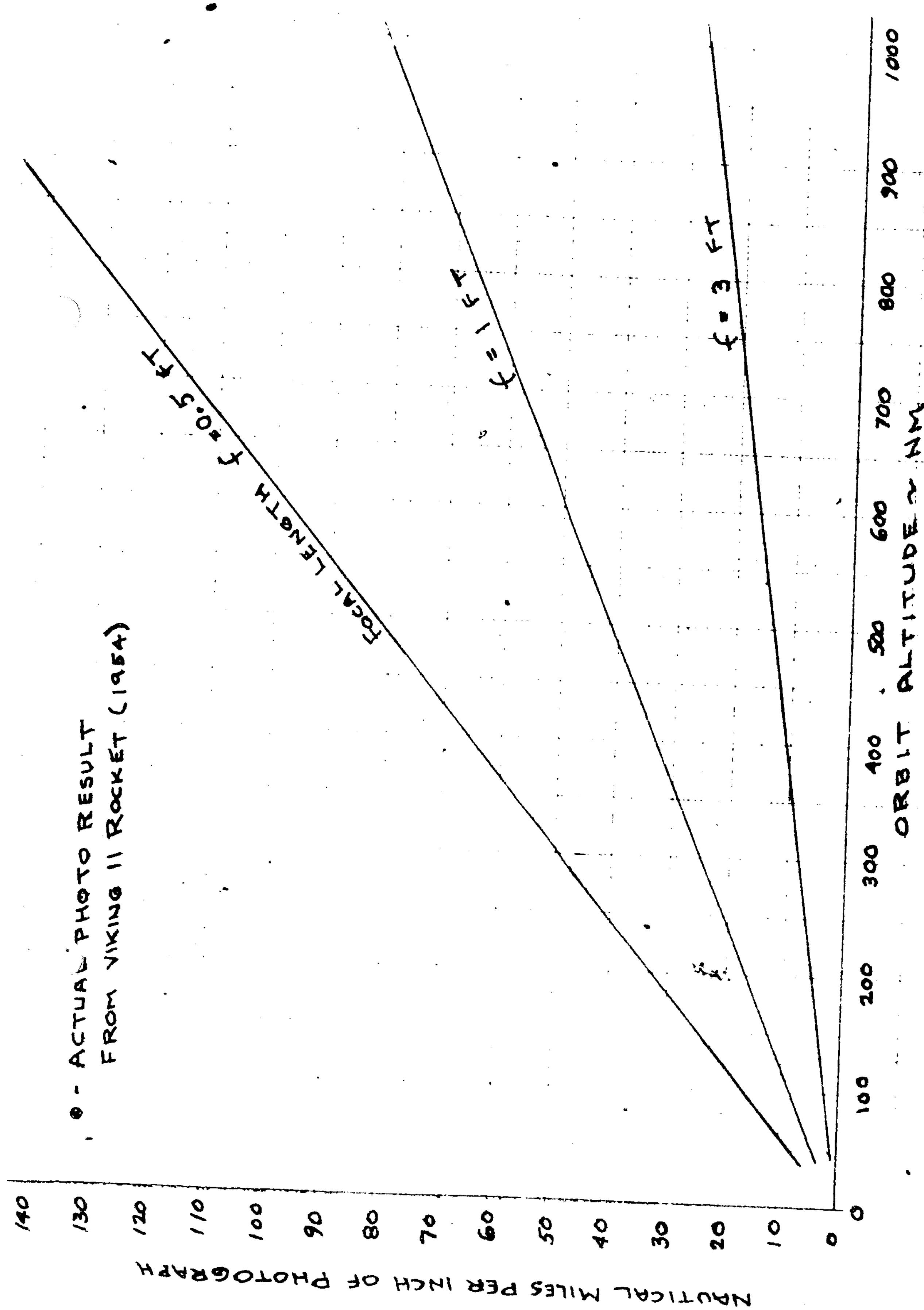

Figure 3-68. Focal Length Influence on Photography from Orbits

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From this formula one would expect to obtain the same ground resolution by trading resolution and scale number. However, this type of reciprocity is never the case either in practice or in theory. If one can trade scale for resolution, the trade should be made in the direction of lower resolution and smaller scale number. There are great differences in the graininess characteristics of different aerial photographic emulsions, and these affect interpretability much more than they influence resolution. Figure 3-69 has been estimated to reflect ground resolutions from various orbit altitudes for current and future photographic capabilities.

Reconnaissance Levels. Operational reconnaissance is expressed by four levels as follows:

Level A - Provides pioneer large-area search. This calls for a system to be operated over areas measured in millions of square miles; for the Earth a scale number of roughly 250,000 K is used (where K is between 1/2 and 2, to account for ignorance and variability of conditions). Ground resolution corresponding to level A is 80 K feet. For space with the lack of atmosphere and other disturbances it is assumed that ground resolution would be better by a factor of 7. Therefore, with a given scale number, ground resolution of 7 times Earth values could be allowed with the same system and would effectively give the same performance. This assumption may not apply for Mars which is believed to have an atmosphere of about 98 percent nitrogen. However, it is assumed adequate for this cursory study in that it is not expected that the Mars atmosphere would have water vapors (clouds), smog or haze that are a part of the Earth's atmosphere.

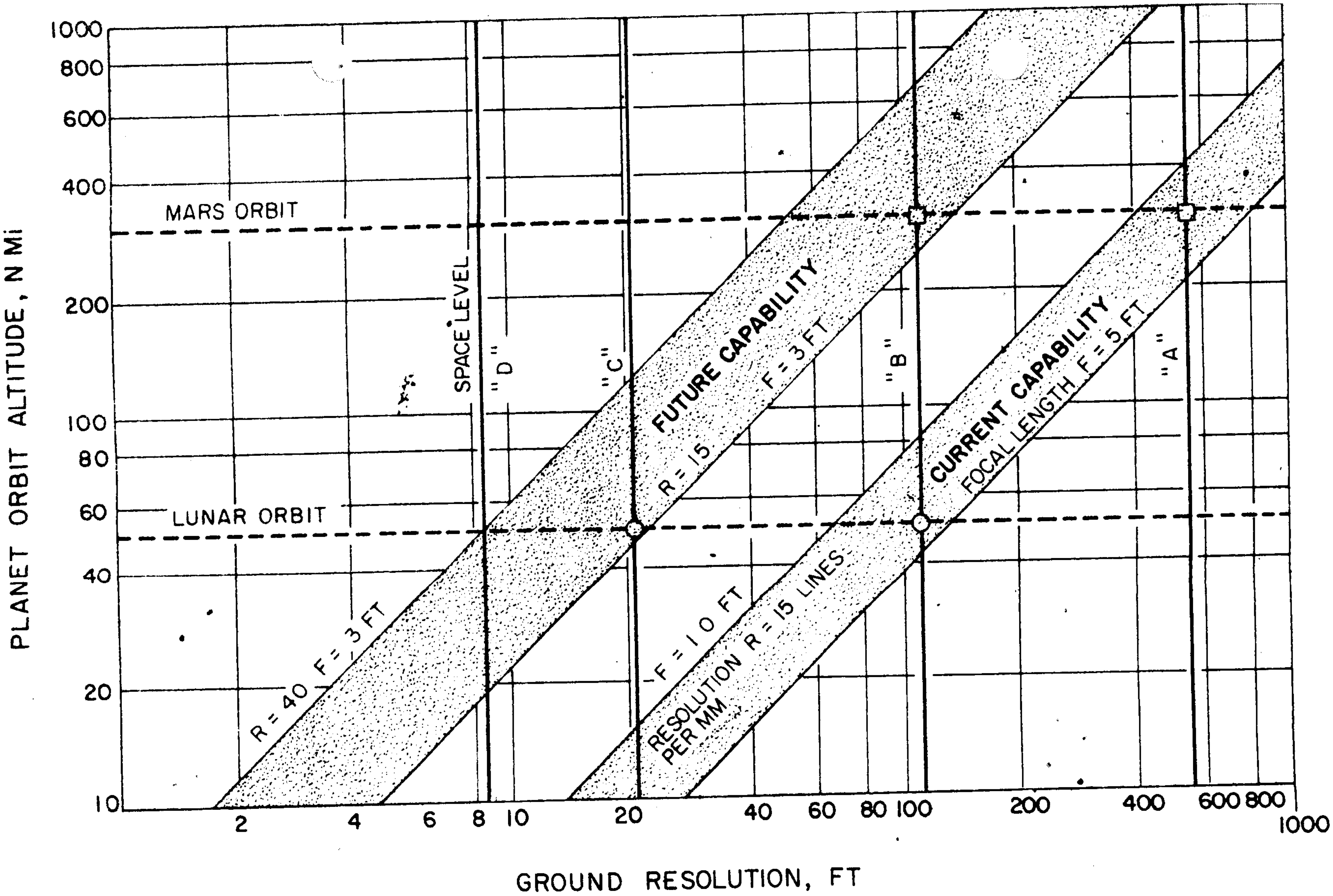

Figure 3-69. Ground Resolutions for Space Orbits Around Planets

Level B - Limited-area search at a scale number of 50,000 K which is useful for areas measured in thousands of square miles. The character of many major installations, facilities, and ground terrain can be detected and identified, aircraft can be seen on airfields, major lines of communications can be found and plotted, and, in general, those items found at level A can be seen more satisfactorily. Ground resolution is about 16 K feet.

Level C - Specific-point-objective photography can be accomplished at a scale level of 10,000 K, and is useful for areas measured in terms of hundreds of square miles. Extremely detailed analyses of sites, airfields, industries, and activities can be made. Topography of terrain such as craters, mountains and hills,foliage, etc., should be well detailed with this level of reconnaissance. Ground resolution is about 3 K feet.

Level D - Technical intelligence objective, at a scale level of 2000 K, is useful for areas which may be less than one square mile. Excellent topography details would result with this level. This requires a ground resolution in the region of 4 to 16 inches.

The above levels are used as guides and are shown in Fig. 3-69 . For initial Mars reconnaissance photography, ground resolutions between levels A and B are adequate for the mission purpose.

Conclusions. As the mission purposed for Mars is essentially reconnaissance of a general nature, space level A coverage is considered satisfactory. This coverage would be a prelude to future missions that would establish ground areas desirable for greater detailed observation and reconnaissance. Considering a Mars orbiting satellite at 300 n mi to provide reconnaissance photography of space level A, we can establish photographic capabilities for the conditions selected. Observation of the planet considering a flat surface would provide a total visual swath width of 1505 n mi. However, useful observation would be only about 600 n mi (Fig. 3-70). To use existing photographic capabilities for the level A coverage, a focal length of 1 foot, 9-1/2-in. film with a resolution of 15 lines/mm is employed. From Fig. 3-68 , 25 n mi/inch of photo would be the scale which would provide a 237 n mi camera swath as shown in Fig. 3-70 . The Mars area per pass for photos would be 1.34×10^6 square n mi, which is about 3 percent of the Mars surface area. As the satellite is orbiting at 6400 knots, and considering that photography would be taken only of the sun-exposed surface, about 30 orbits and 5 days would be required to cover the entire Mars surface area. This would require about 2640 feet of 9-1/2-in. film at 93 lb film weight. Other requirements could be determined dependent upon orbit conditions and space levels desired using Fig. 3-68 and 3-69 , but is beyond the scope of this report. However, this limited investigation illustrates the feasibility of the selected Mars orbit (300 n mi) assuming that the space level A reconnaissance level is adequate for the space mission purpose.

Instantaneous Impulse Maneuver

The next phase in the Mars mission requiring propulsion is that of establishing an orbit about Mars. In establishing the Martian orbit, the primary objective is to affect capture into some satellite orbit

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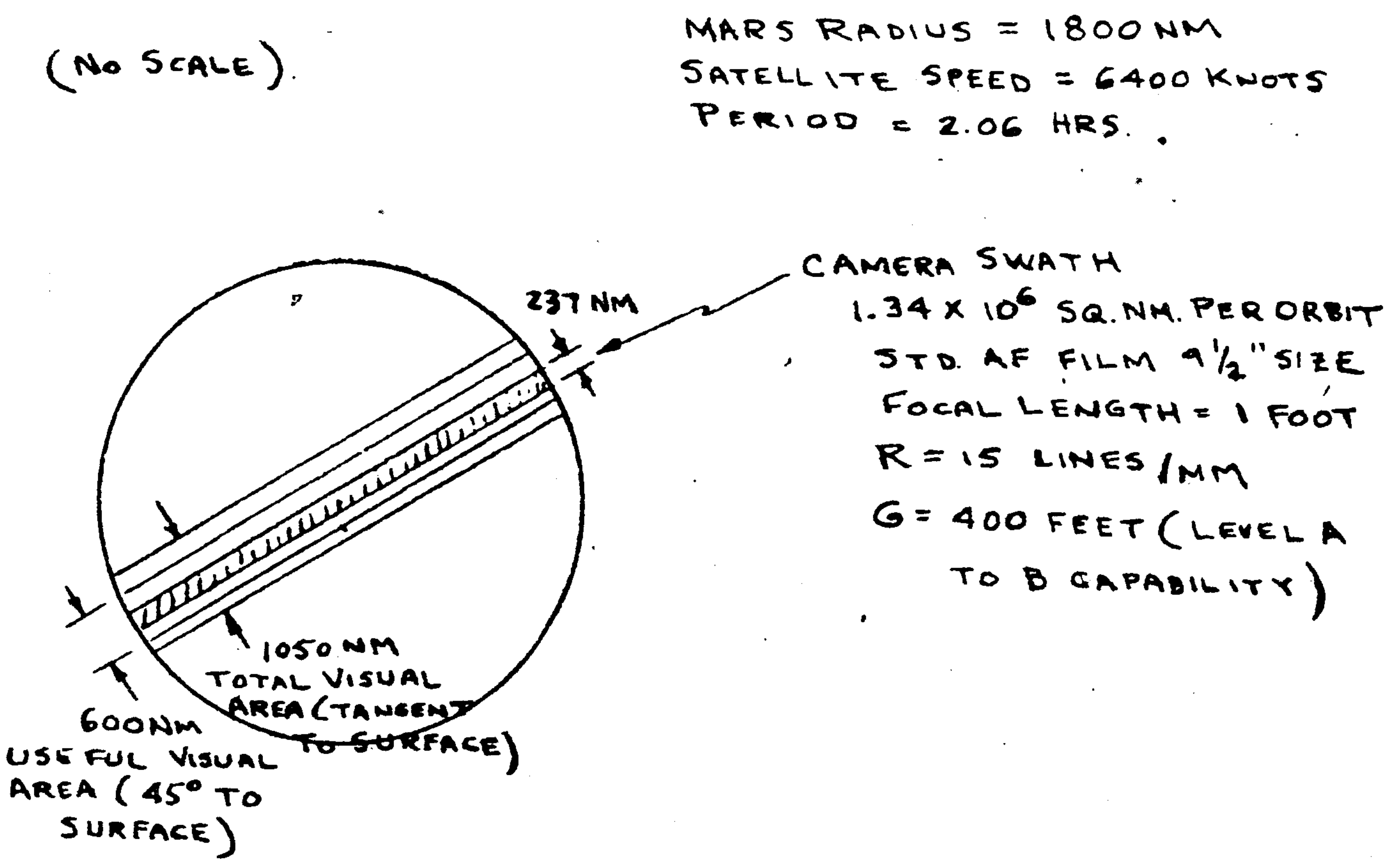

Figure 3-70. 300 n mi Mars Orbit

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to prevent the vehicle bypassing Mars. With this philosophy, the inclination of the resulting Martian satellite orbit is not considered an important parameter. If the inclination between the Martian equatorial plane and the resulting orbital plane is incompatible with mission goals, propulsion can be used to change the plane to a selected plane. The orbit establishment analyses have been conducted without regard to the inclination of the resulting orbit.

An ideal situation for establishing the Martian orbit assumes zero burning time or instantaneous change of the velocity vector. In this ideal case there would be no losses. The results of the ideal approximation to an actual propulsion phase indicate trends to be expected from finite burning periods. Figure 3-71 , 3-72 , and 3-73 have been constructed for this ideal case of instantaneous change in the velocity vector. Each figure is constructed for a different hyperbolic approach velocity to Mars. The ideal velocity increment supplied by the propulsion system for establishing an orbit, or for effecting capture, is a function of the orbit height above the Martian surface.

The vehicle approaches Mars from infinity with a hyperbolic excess velocity; as it comes nearer to Mars the velocity of the vehicle (with respect to Martian coordinate system) increases until a maximum velocity is attained when the vehicle reaches the distance of closest approach (vertex) on its hyperbolic trajectory. The ideal velocity increment of Fig. 3-71 , 3-72 and 3-73 is applied at the vertex. The velocity increment for establishing a circular orbit at any Martian altitude with this instantaneous change in velocity applied at the vertex can be obtained by taking the difference between the entry velocity curve and the circular orbit velocity curve.

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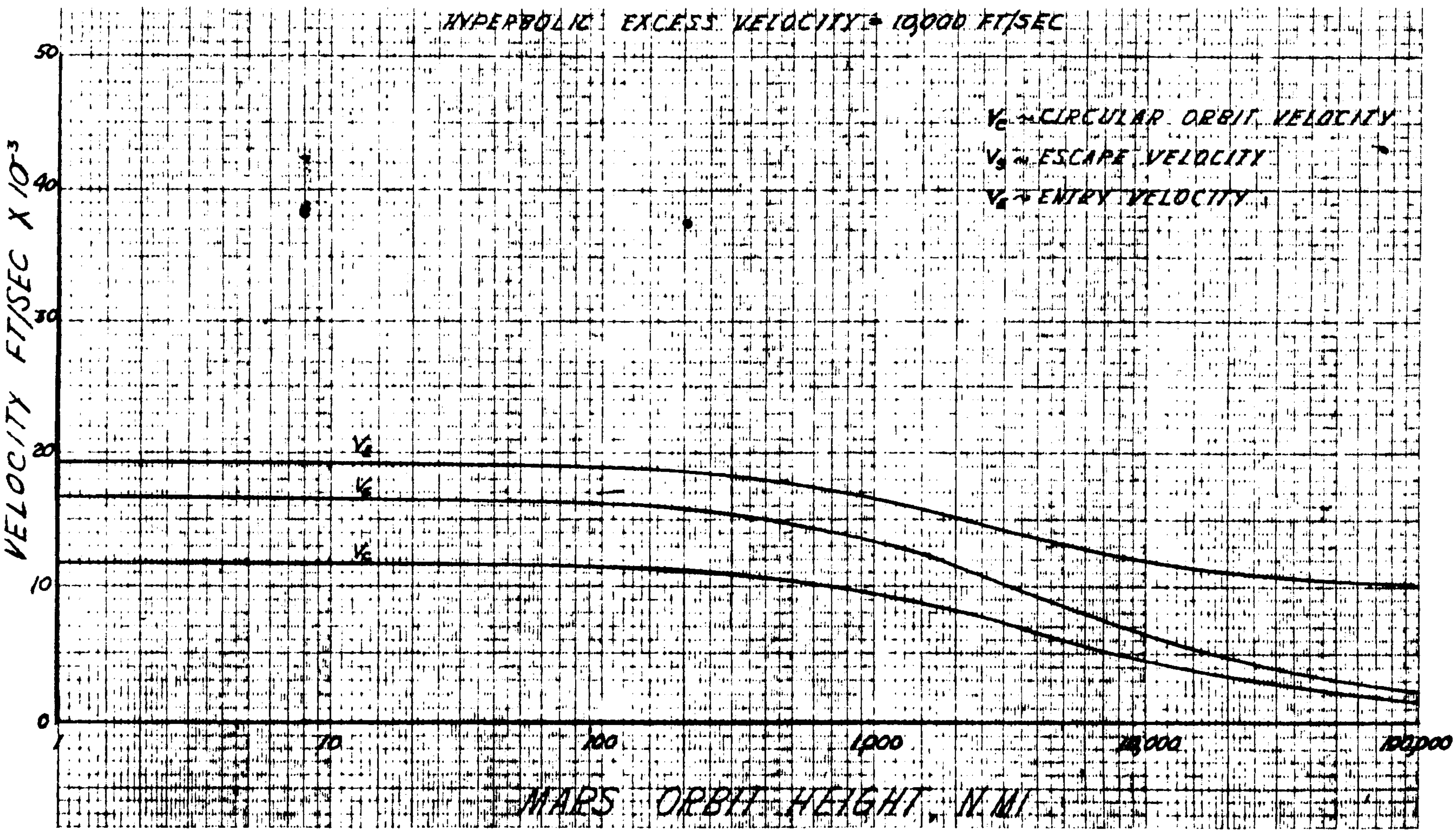

Figure 3-71. Mars Hyperbolic Approach
Hyperbolic Excess Velocity = 10,000 ft/sec

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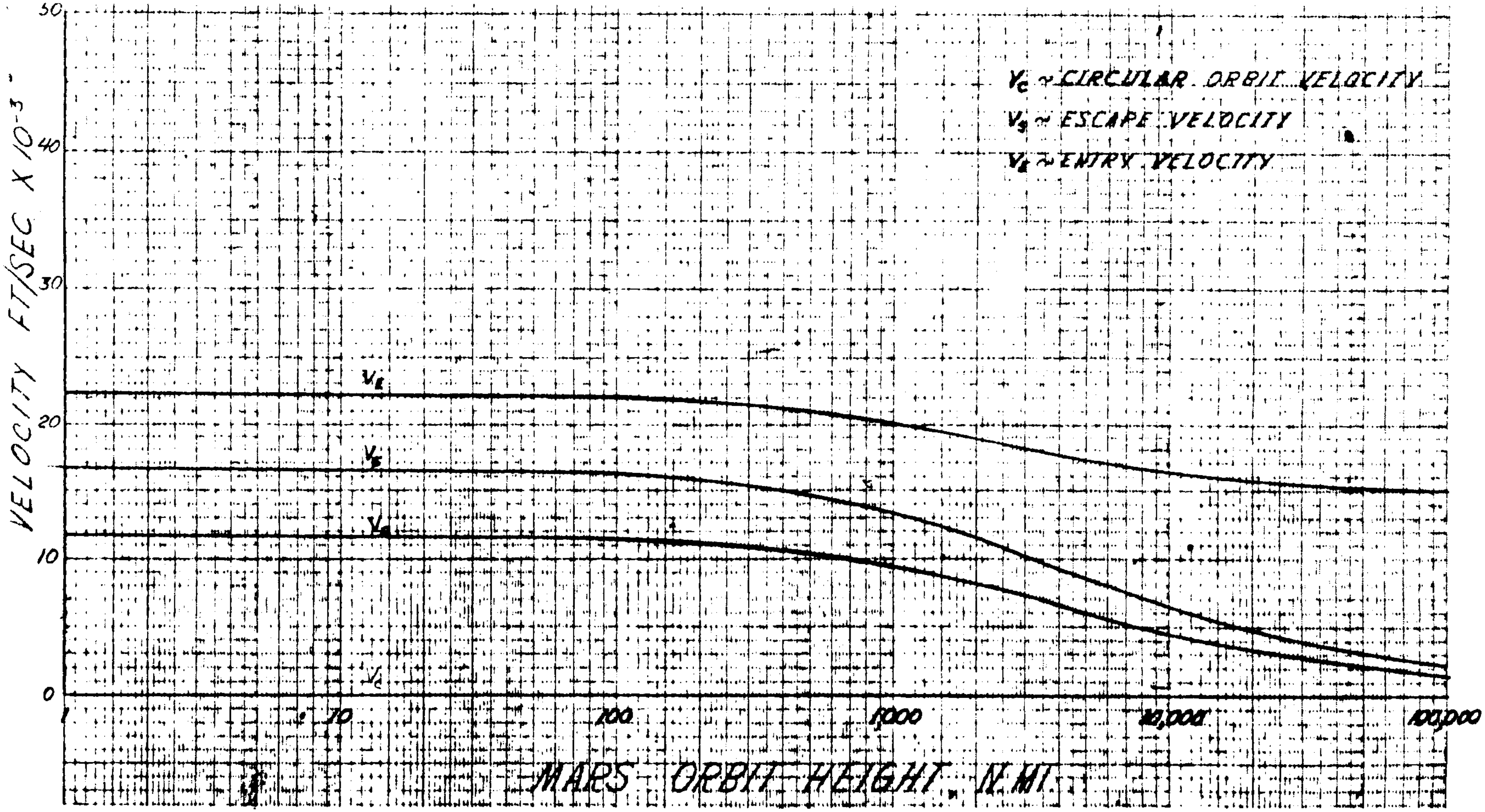

Figure 3-72. Mars Hyperbolic Approach, Hyperbolic Excess Velocity = 15,000 ft/sec

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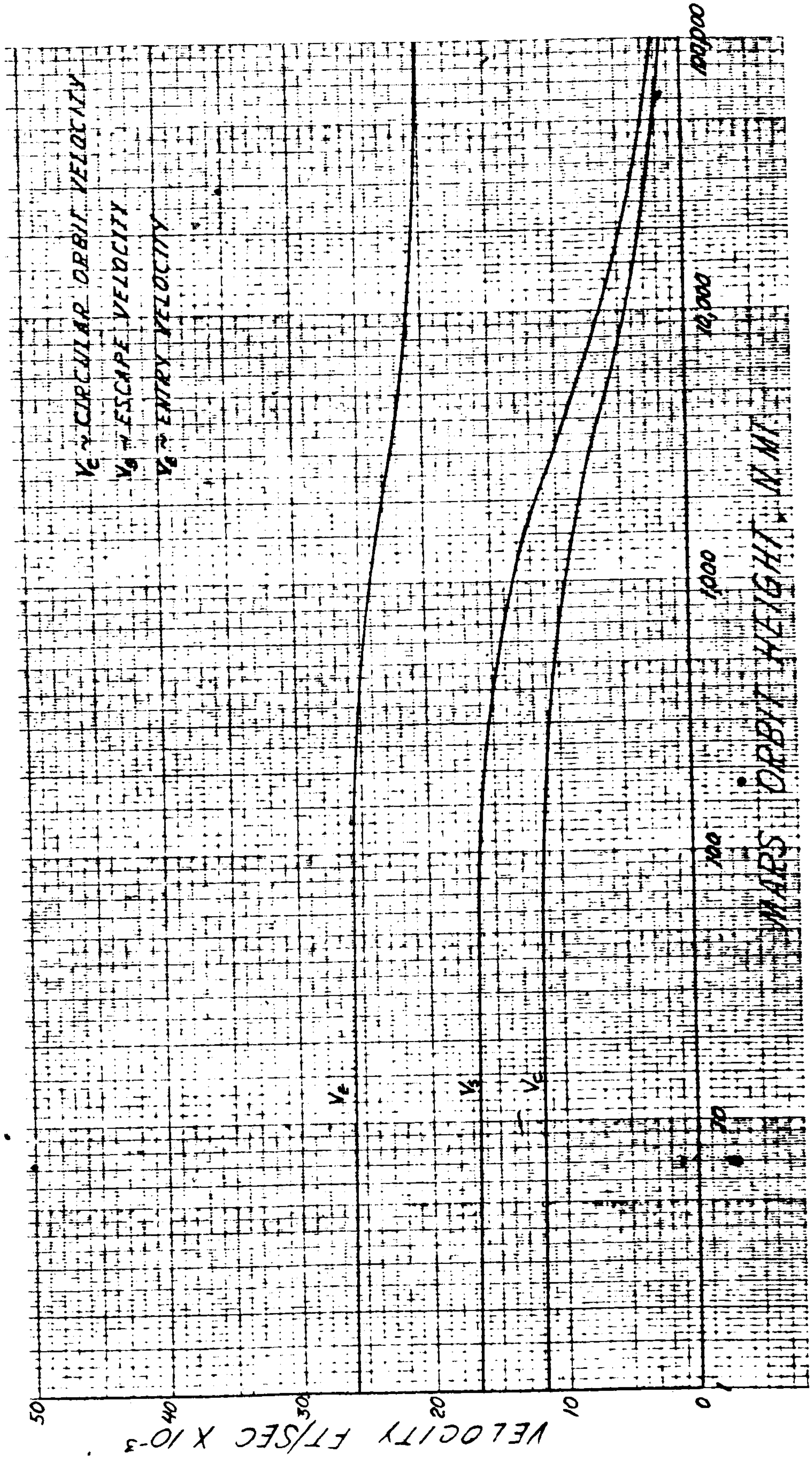

Figure 3-73. Mars Hyperbolic Approach, Hyperbolic Excess Velocity=20,000 ft/sec

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The preceding figures have shown the ideal velocity increment for establishing a circular orbit about Mars for orbital altitude varying from 1 n mi up to 100,000 n mi. For the second phase of this present NASA study, a 300 n mi orbit has been selected for recommendation. Having selected the 300 n mi Martian orbit as the nominal case, Fig. 3-74 presents the instantaneous velocity increment for changing the hyperbolic trajectory into a circular orbit. This ΔV is plotted as a function of the hyperbolic approach velocity.

In conjunction with the analysis for an instantaneous impulse method of orbit establishment about Mars, the asymptotic approach distance of the vehicle must be controlled to achieve an altitude of 300 n mi at the vertex (Fig. 3-75). For various launch dates, the vehicle approaches Mars with varying hyperbolic excess velocities. The asymptotic approach distance must be varied as a function of the hyperbolic approach velocity (Fig. 3-76) to effect a common vertex at 300 n mi altitude. As the hyperbolic approach velocity decreases, the asymptotic approach distance approaches infinity. As the hyperbolic approach velocity increases, the curve of Fig. 3-76 becomes asymptotic to the vertex distance (300 n mi altitude).

The asymptotic approach distance can be controlled during the midcourse correction phase, if not obtainable during Earth orbit departure phase, by varying the radial miss distance. The radial miss distance is defined as the closest distance between the vehicle heliocentric trajectory and a point mass planet (which does not perturb the vehicle's heliocentric trajectory). Figure 3-77 shows how the asymptotic approach distance (d) varies with the radial miss distance between the vehicle and the point mass planet.

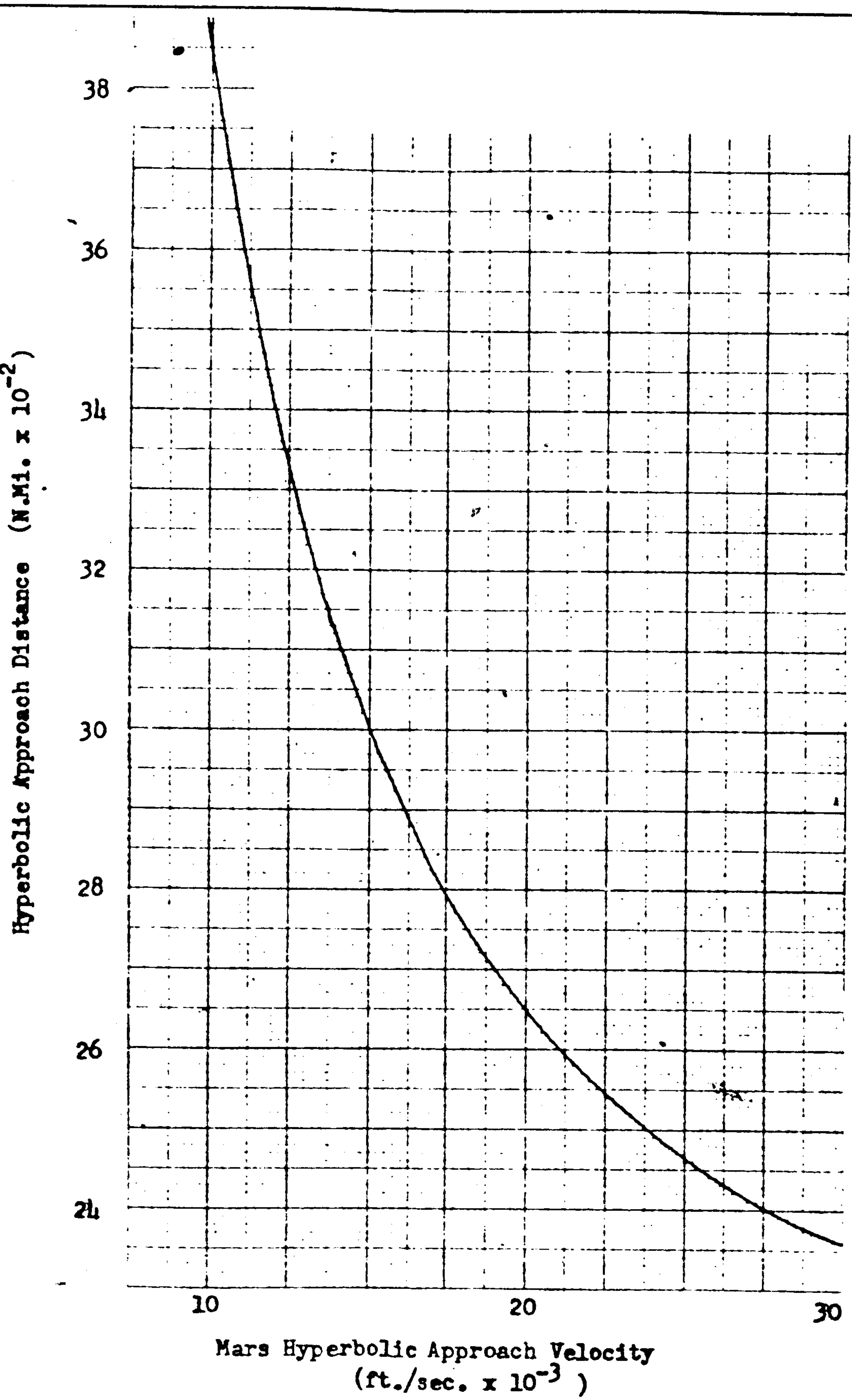

Figure 3-76. Parameters Affecting A 300 n mi Hyperbolic Grazing Encounter Above the Surface of Mars

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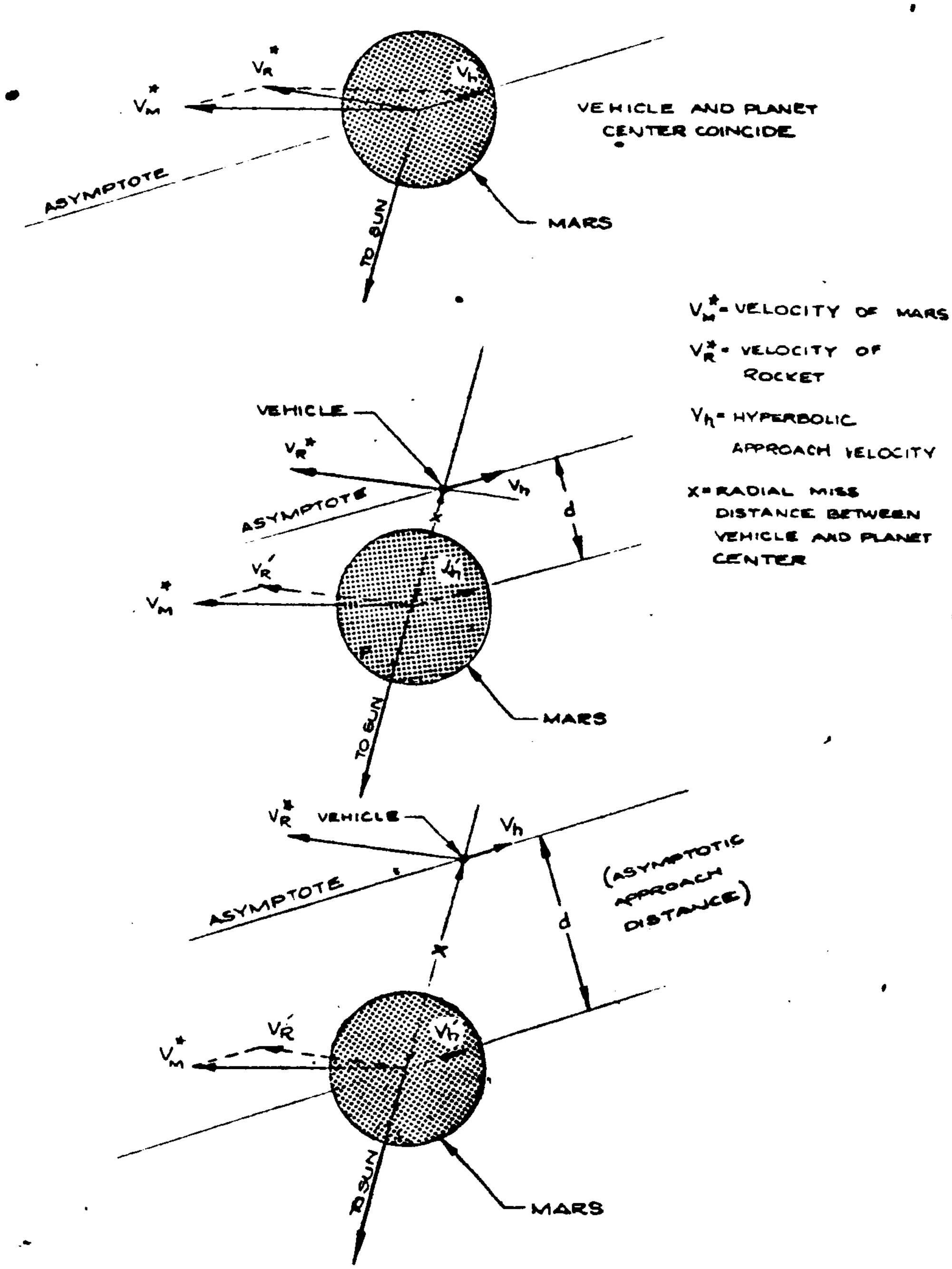

*Heliocentric Velocities

Figure 3-77. Variation in Asymptotic Approach Distance with Radial Miss Distance

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Simulated Trajectory Maneuvers

The preceding discussion has been for establishing a Martian orbit with an instantaneous impulse to change the vehicle velocity vector. This analysis provides the basis for additional analyses to investigate techniques of orbit establishment for vehicles with propulsion systems which burn for finite periods of time.

As stated previously, the Mars hyperbolic approach velocity of the space vehicle varies with the launch date. The hyperbolic approach velocity for an interval of launch dates in the 1964 period selected for study varies from approximately 13,700 to approximately 18,800 feet per second. This variation in hyperbolic excess velocity has an important effect on the Mars orbit establishment maneuver.

Direct Final Orbit Establishment. Several propulsion maneuvers have been investigated for changing the space vehicle hyperbolic trajectory directly to a circular Martian orbit. For circular orbit establishment, constant thrust and constant thrust-to-weight propulsion programs have been evaluated. The maneuvers for circular orbits have been simulated on digital computers by flying the vehicle backwards out of the circular orbit, using a negative mass flowrate. The powered phase of the maneuver terminates when the vehicle attains an energy level associated with the hyperbolic excess velocity. From the nomograph of Fig. 3-61 , the weight of the vehicle in the circular orbit can be obtained for each hyperbolic approach velocity, since the corresponding gross weight of the second stage (Fig. 3-63) is known. The altitude at which the vehicle reaches the energy level is the altitude for initiating retrothrust when establishing a circular orbit. From other trajectory parameters at this energy level termination, the asymptotic approach distance can be calculated from the momentum equation.

For a constant tangential retrothrust propulsion maneuver for establishing a Mars 300 n mi circular orbit, the entry corridor parameters are presented in Fig. 3-78 . The asymptotic approach distance which must be achieved for the 200 day transfer mission and the selected interval of departure dates during the 1964 period varies between approximately 2800 n mi and 3200 n mi. It is the function of the midcourse propulsion and guidance-control systems to establish these entry corridor requirements. Figure 3-79 depicts the variation in the entry corridor geometry for this orbit establishment mission.

For a constant thrust-to-weight vehicle (thrust-to-Earth weight equal to 0.39)* using a thrust opposing velocity maneuver, the retrothrust burning time, the asymptotic approach distance, and the altitude for retrothrust initiation are shown in Fig. 3-80 . These parameters are functions of the hyperbolic approach velocity and are relative to establishing a 300 n mi circular orbit.

Intermediate Orbit Establishment. For establishing the intermediate orbit about Mars it may simplify midcourse guidance requirements if a common asymptotic approach distance (Fig. 3-81) could be used regardless of hyperbolic excess approach velocity. Additionally, it would greatly simplify the problem if a common altitude (also independent of the hyperbolic approach velocity) could be used for beginning of the retrothrust maneuver. Granted, these simplifications would result in intermediate orbits different from the final 300 n mi circular orbit; but, this intermediate orbit could be corrected by some other propulsion phase to a 300 n mi orbit.

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*A 0.39 thrust-to-Earth weight ratio corresponds to approximately a 1.0 thrust-to-Mars weight ratio.

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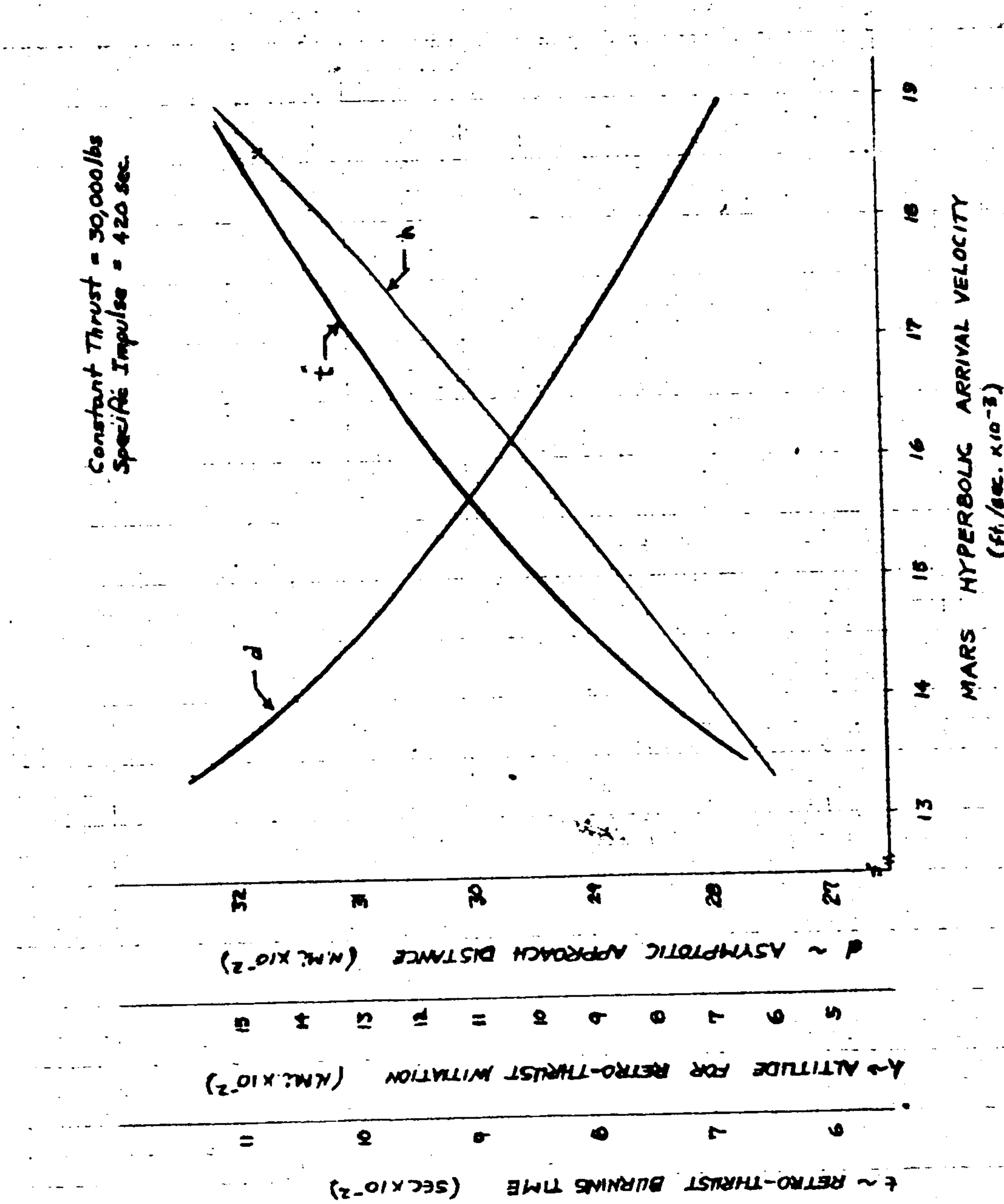

Figure 3-78. 200 Day Mars Transfer

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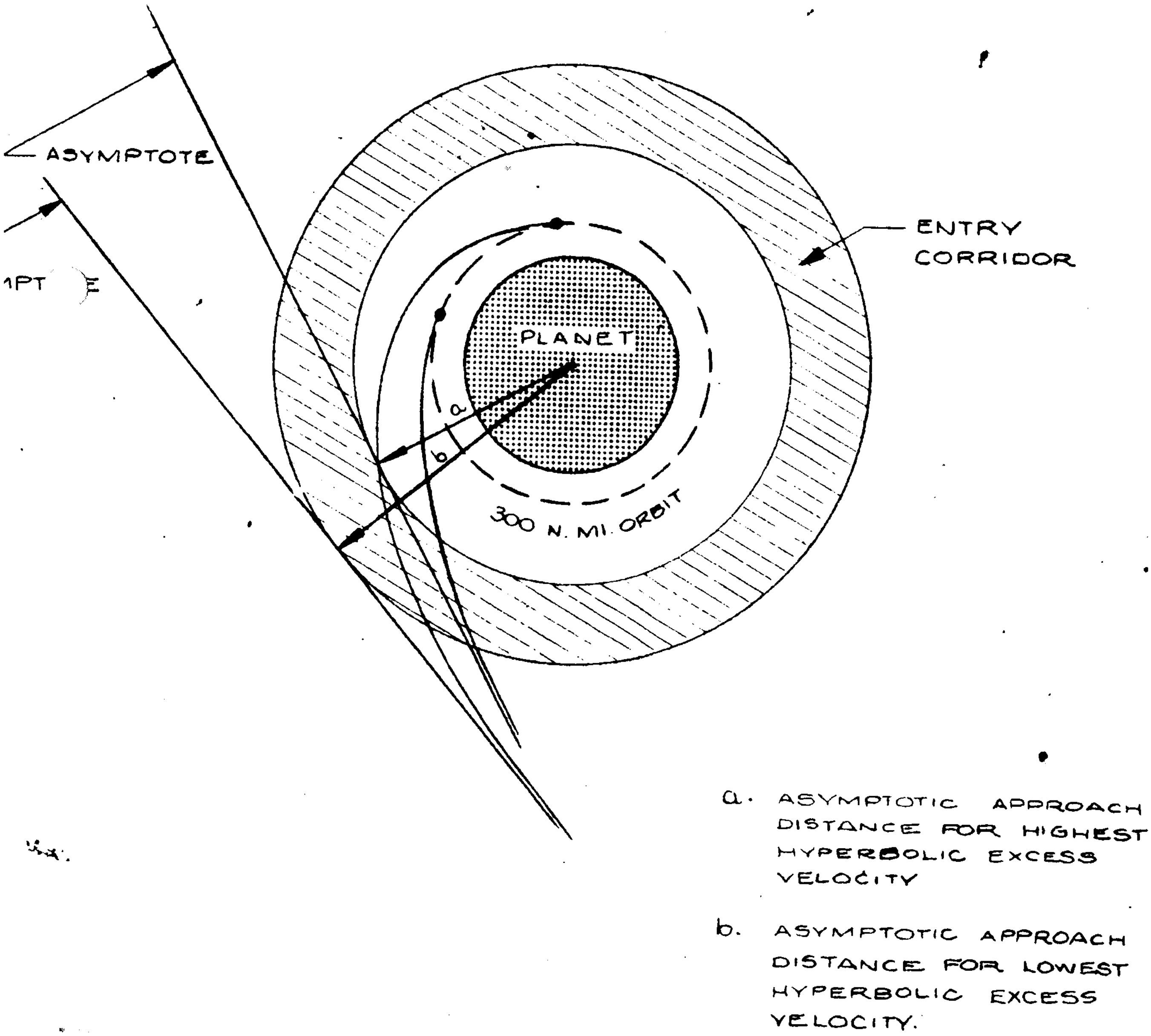

Figure 3-79. Mars Orbit Establishment Maneuver

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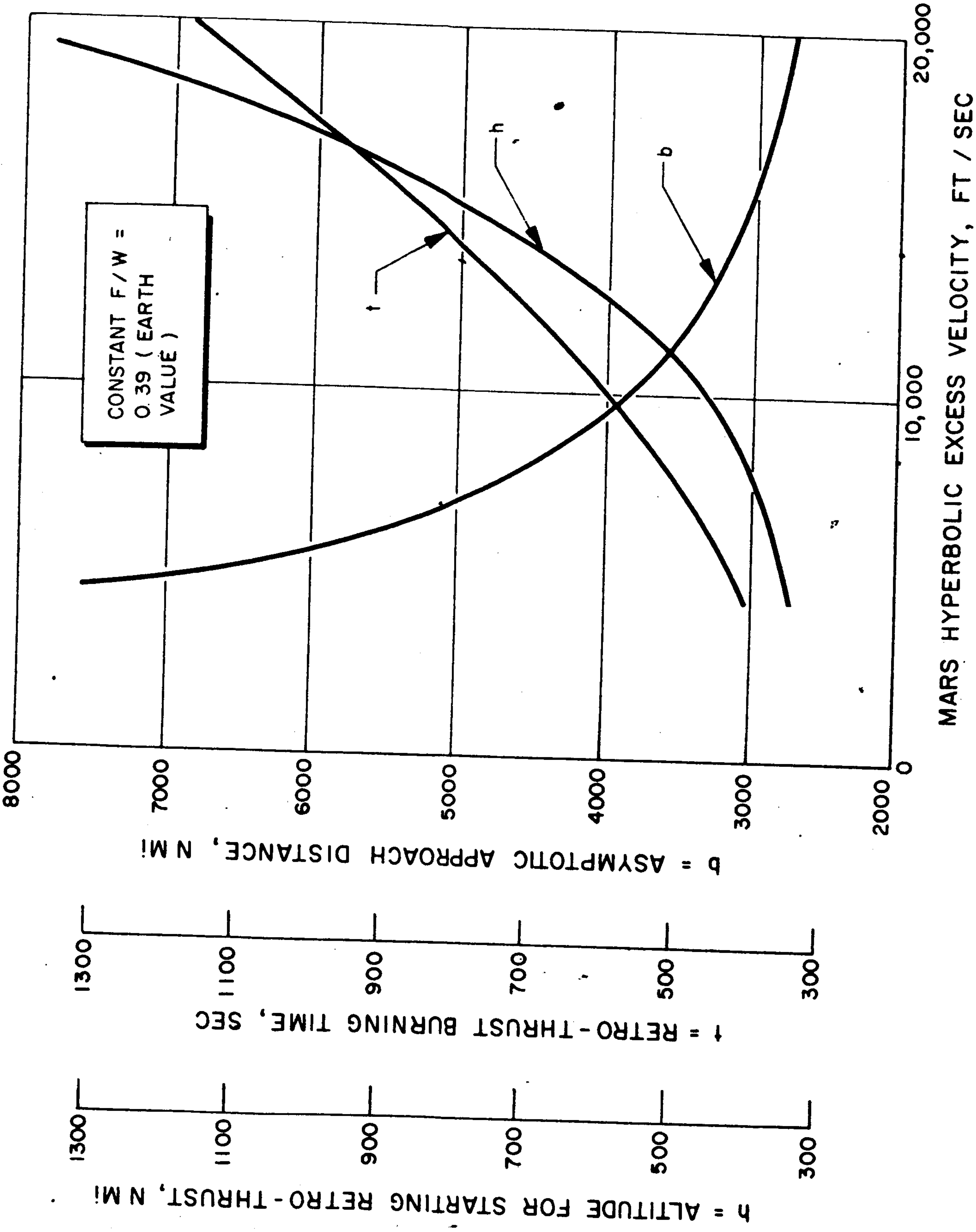

Figure 3-80. Mars Transfer

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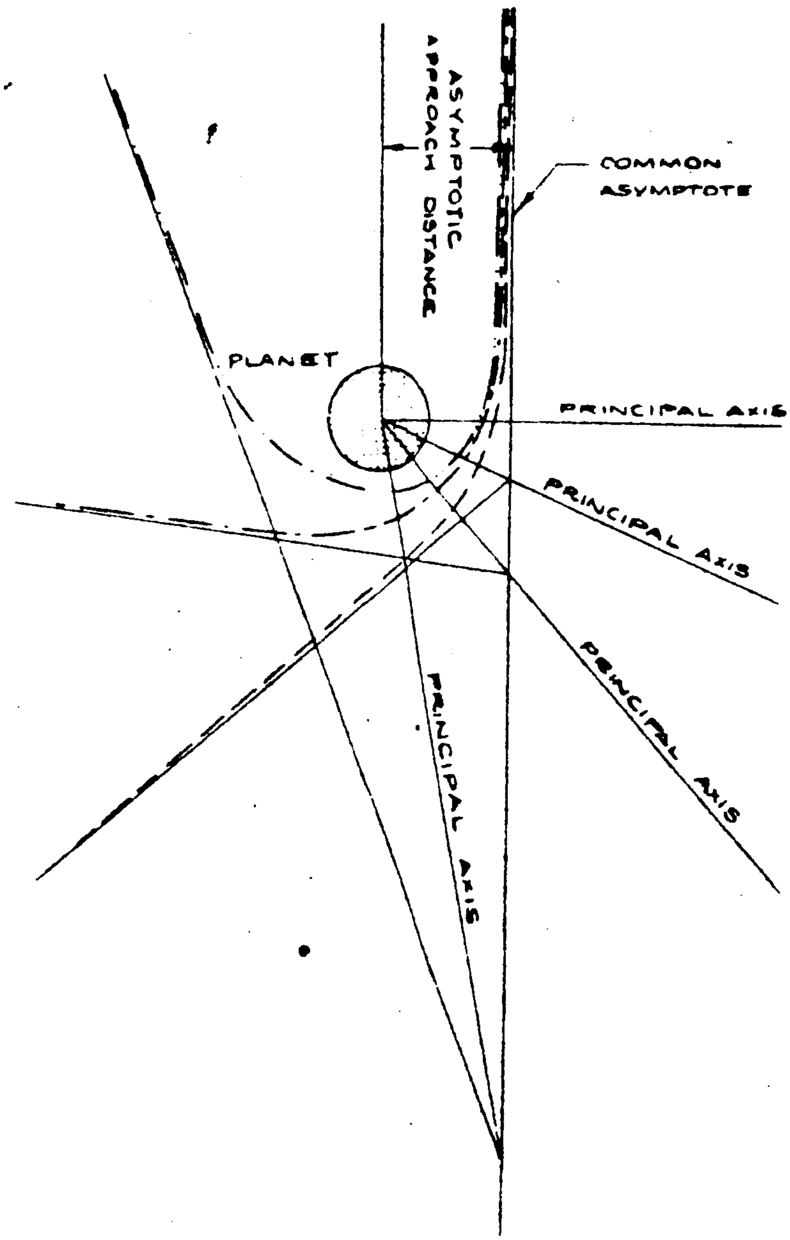

Figure 3-81. Hyperbolic Approach Trajectories With A Common Asymptotic Approach Distance

An analysis was performed to determine whether or not it is feasible to use a tangentially applied constant thrust-to-weight program to establish an orbit employing the concept of a common asymptotic approach distance and a selected initial altitude for beginning the retrothrust maneuver. In addition, the propulsion system would deliver a total impulse which is a function of the hyperbolic approach velocity. The total impulse would correspond to that required for maneuvers described by Fig. 3-80. If the asymptotic distance from Fig. 3-80 for the 18,800 ft/sec hyperbolic excess velocity trajectory were selected as that to be used for all trajectories, those trajectories with low hyperbolic excess velocity (in the order of 13,700 ft/sec) resulted in orbits which intercept the surface of the planet Mars. This intersection occurs even with no retrothrust applied to the vehicle. Thus, the altitude for retrothrust initiation becomes meaningless. Perhaps with additional analysis an asymptotic distance, an altitude and a total impulse could be determined suitable for these simplifications with a constant thrust-to-weight propulsion maneuver. However, from the results of the limited analyses conducted for constant thrust-to-weight maneuvers, the simplifications must be ruled out.

It is recommended on the basis of analysis accomplished for constant thrust-to-weight propulsion systems that the asymptotic approach distance and the altitude for the initiation of the retrothrust maneuver be programed into the guidance and control system for each particular hyperbolic approach velocity.

For the concept of establishing a Mars orbit by using a common asymptotic approach distance and a selected initial altitude for the beginning of a retrothrust maneuver, a constant thrust program was investigated. For the constant thrust analyses the total impulse delivered by the propulsion system was selected to be a function of the hyperbolic approach velocity as described by Fig. 3-78.

As stated in the discussion for constant thrust-to-weight propulsion programs, selecting the asymptotic approach distance corresponding to the high hyperbolic approach velocity (18,800 ft/sec), results in trajectories for low hyperbolic approach velocities that intercept the surface of the planet even without retrothrust being applied to the vehicle.

For a constant thrust propulsion maneuver in which the retrothrust is applied in opposition to the velocity, an investigation was conducted using the asymptotic approach distance of the minimum hyperbolic approach velocity (13,700 ft/sec). The altitude for retrothrust initiation was selected as that value which would result in establishment of a 300 n mi circular orbit (for a hyperbolic approach velocity of 13,700 ft/sec) using the described propulsion maneuver.

As the analysis progressed the simplification of using a common altitude for initiation of retrothrust was deleted. Retaining the common asymptotic approach distance, space vehicles with large hyperbolic approach velocities (18,800 ft/sec) have distances of closest approach (altitude of the hyperbola vertex) higher than the selected altitude for retrothrust initiation. No analysis was performed to determine the effects of increasing the altitude above the selected value. On the basis of the analysis conducted, it was concluded that a common altitude simplification could not be used for initiating the retrothrust maneuver independent of the hyperbolic approach velocities occurring within the range considered.

It was then decided that the altitude for beginning the retrothrust maneuver should be a function of the hyperbolic approach velocity. The altitude for each hyperbolic approach velocity was selected as presented

[redacted]

in a previous section and plotted in Fig. 3-78 . The results of this method of orbit establishment appear favorable for affecting an intermediate orbit about Mars.

The periapsis and apoapsis distances of the resultant intermediate orbits are shown in Fig. 3-82 along with the altitude and retrothrust burning time for this technique of orbit establishment. The periapsis distance varies between approximately 250 n mi and 400 n mi. However, the apoapsis distance varies between 300 n mi and approximately 1000 n mi as a result of this simplification.

For the selected asymptotic approach distance, and the dependency of the total impulse upon Mars hyperbolic approach velocity, the largest deviation between the intermediate orbit and final orbit parameters occurs for the highest hyperbolic approach velocity (18,800 ft/sec) of the 1-month period of Earth orbit departure dates. Thus, the maximum total impulse required of the second stage will be the sum of the total impulse for establishing the intermediate orbit when the vehicle has a hyperbolic approach velocity of 18,800 ft/sec and the total impulse to change the resultant intermediate orbit into the final orbit. The minimum second stage total impulse requirement will occur when the hyperbolic approach velocity of the 1-month period is a minimum (13,700 ft/sec). At this approach velocity the intermediate orbit actually corresponds to the final orbit (Fig. 3-82).

For the constant retrothrust maneuver, additional analyses could be conducted to determine the variation in resultant orbital parameters with changes in the delivered total impulse for a particular hyperbolic approach velocity. Some total impulse different from the assumed value may be determined that would give better intermediate orbital parameters than those of Fig. 3-82 .

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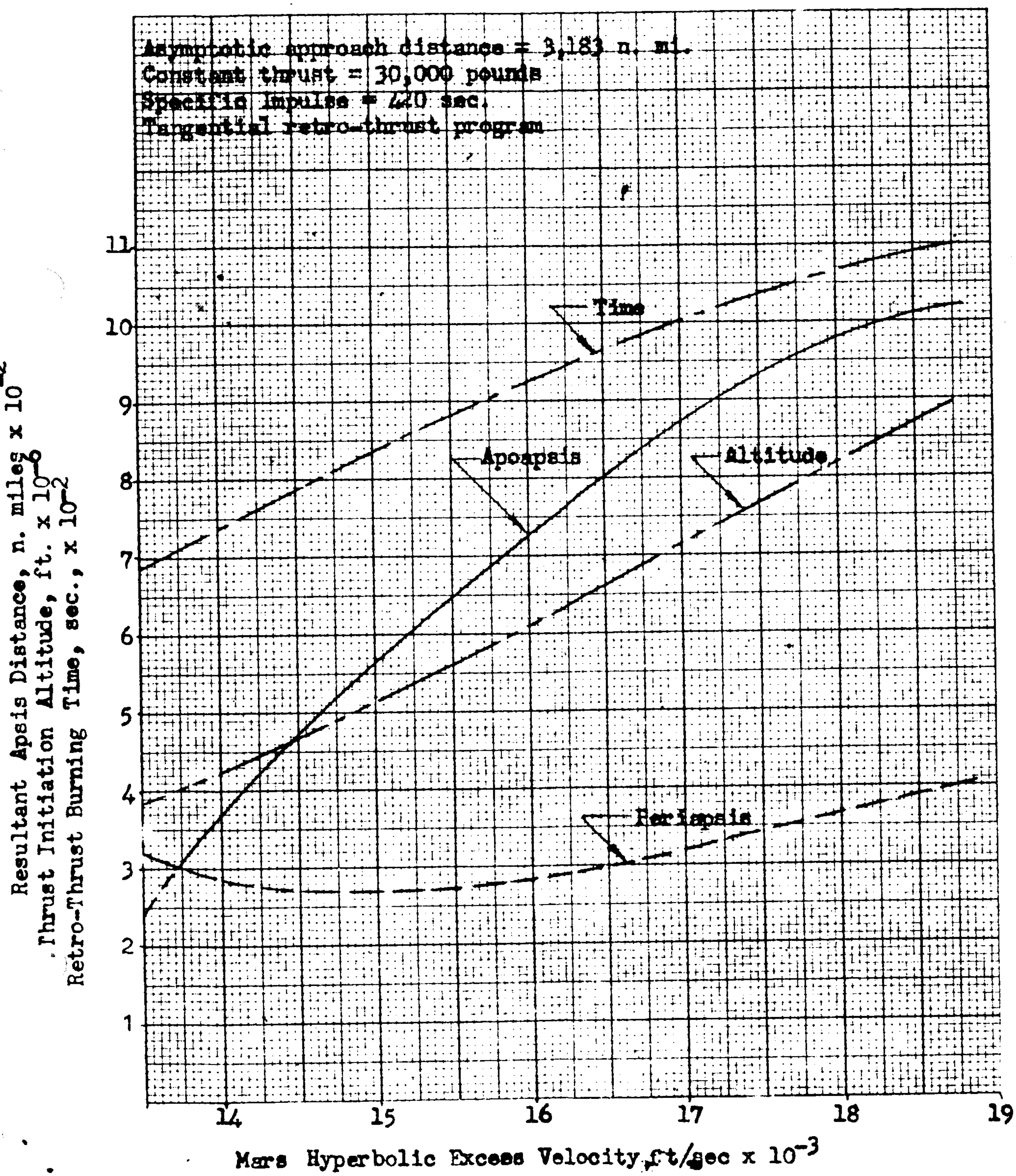

Figure 3-82. 200 Days Mars Transfer Mission

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If this type of orbit establishment propulsion maneuver is used, the midcourse propulsion/guidance systems must establish the asymptotic approach distance. The terminal guidance and control system for an intermediate orbit establishment would consist chiefly of a radar altimeter to determine the altitude for initiation of the retrothrust maneuver.

Intermediate-to-Final Orbit Maneuver. In previous paragraphs, discussion has been presented pertaining to establishing an intermediate orbit which later could be corrected to a 300 n mi circular orbit. Because of the inaccuracies in the Mars orbit establishment maneuver and because of the variations in the intermediate orbits due to the change in the Mars approach hyperbola, the actual orbit established may differ considerably from the desired 300 n mi orbit. It is therefore desired to correct this intermediate orbit to the 300 n mi circular orbit recommended in this study.

Figure 3-83 represents the impulsive ideal velocity requirement to transfer from a Mars elliptical orbit to a 300 n mi circular orbit. A minimum-energy two-impulse transfer was used in evaluating the velocity requirement. In Fig. 3-83 the ideal velocity requirement is plotted vs the ellipse perifocus distance for various apofocus distances.

To use this figure read up from the orbit perifocus distance of the intermediate orbit to the line corresponding to the apofocus distance of the intermediate orbit; then, read across to determine the total impulsive ideal velocity increment required to transfer from the intermediate orbit to the 300 n mi circular orbit.

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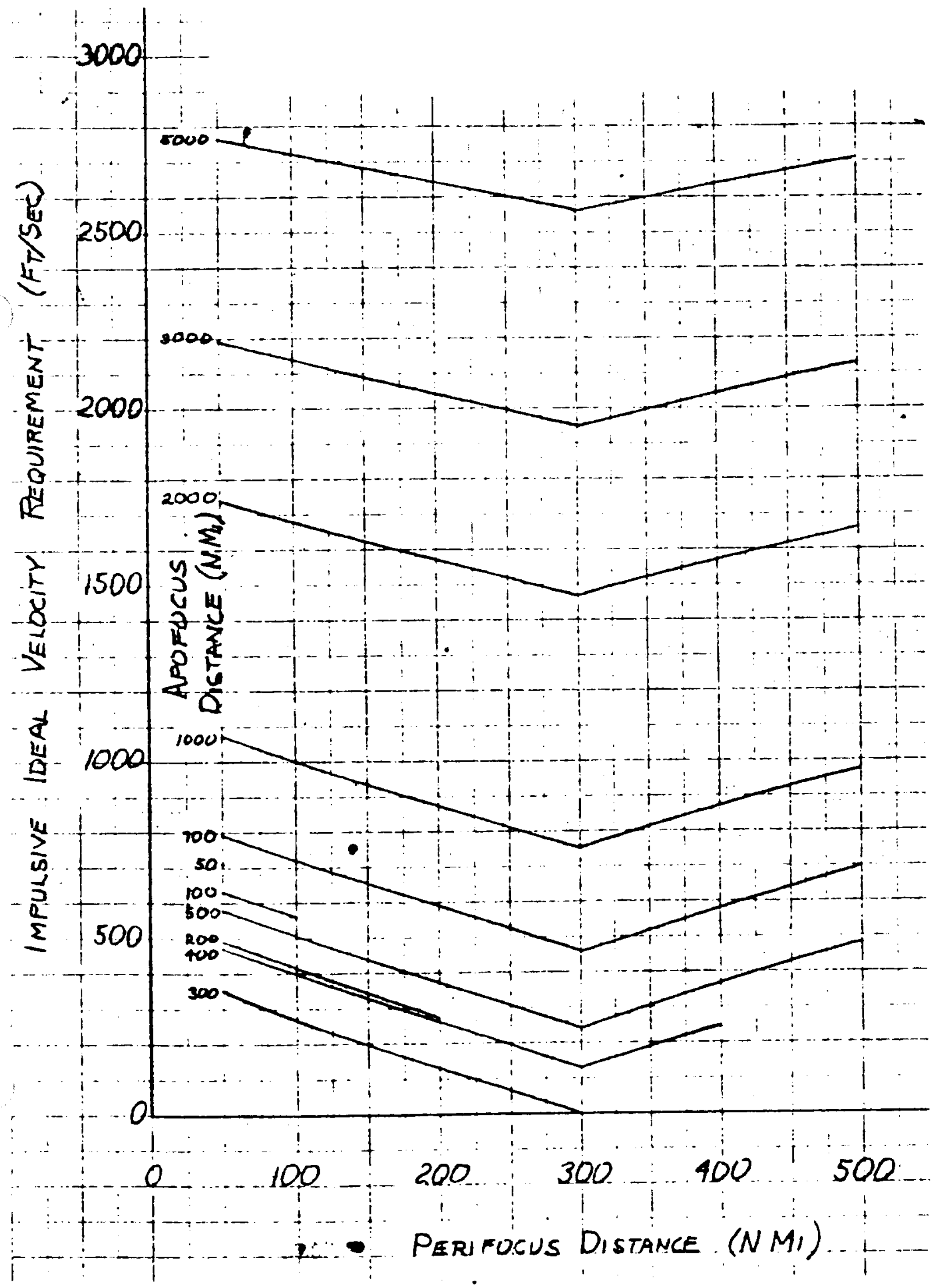


Figure 3-83. Velocity Requirement to Change Mars Elliptical Orbit to 300 n mi Circular Orbit

[redacted]

No integrated trajectory propulsion maneuver analysis was conducted to determine the increase in velocity increments for actual propulsion systems performing this change from the intermediate orbit to the final orbit. However, analyses conducted for Earth orbit changes, indicate that for a Mars orbit change propulsion maneuver performed with a thrust-to-(Earth) weight greater than approximately 0.2, the actual velocity increment will not be appreciably larger than the ideal velocity increment.

The largest velocity increment to change the intermediate orbit to the final orbit occurs for the highest hyperbolic approach velocity of the range considered (13,700 to 18,00 ft/sec). From Fig. 3-82, the intermediate orbit for the 18,800 ft/sec hyperbolic approach velocity has a 400 n mi periapsis and a 10,100 n mi apoapsis. The velocity increment (Fig. 3-83) to change this orbit to the 300 n mi circular orbit is approximately 875 ft/sec. For performing the Mars mission with an intermediate orbit concept, the second stage propellant capacity must be large enough to provide this additional velocity increment.

VARIATIONS IN PROPULSION SYSTEM, VEHICLE AND MANEUVER PARAMETERS

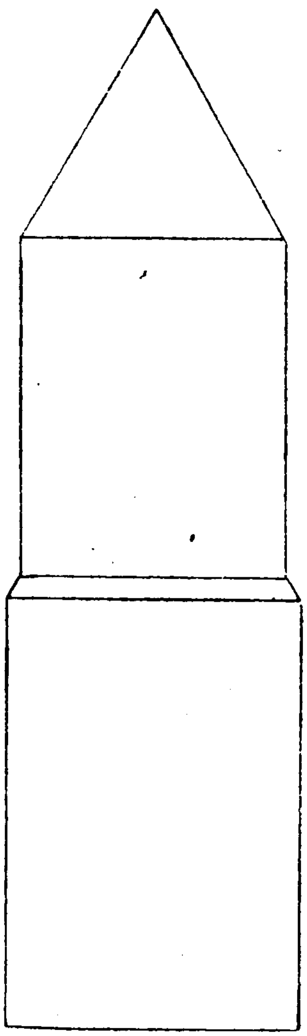
Many factors affect the payload weight that can be placed in a Mars 300 n mi circular orbit for a vehicle with a fixed initial gross weight. Some of the propulsion system parameters that affect payload are the specific impulse of the stages of the vehicle; the thrust level of the stages; and the structure weight or inert weight of the stages. The significance of these parameters will be analyzed in more detail for resultant variation in the payload of a space vehicle with an initial gross weight of 354,000 lb. This vehicle corresponds to the payload placed in a 300 n mi Earth orbit by a Nova H-6 vehicle.

In a previous section, a constant thrust, retrothrust program for direct orbit establishment was described (Fig. 3-78). The asymptotic approach distance, and the altitude for thrust initiation were functions of the hyperbolic approach velocity. This type of trajectory has been used for the analyses of the effect of parametric variations on the establishment of a Mars 300 n mi circular orbit. Although the analyses are for a direct orbit establishment maneuver, and an intermediate orbit maneuver has been recommended for this mission, it is believed that these studies for direct orbit establishment will be indicative of results to be expected for the recommended maneuver.

Staging

For evaluating some of the propulsion system parameters, the 354,000 lb vehicle will be considered as a two stage vehicle (Fig. 3-84): one stage for escaping the Earth orbit and establishing the heliocentric transfer trajectory; and a second stage for establishing the Mars orbit. This assumption does not prevent inclusion of another stage (or even parallel staging) as a part of the payload of the second stage. A restart of the second stage (or an additional stage) required to change the intermediate orbit into the final orbit will reduce the indicated payload into the final orbit by the weight of this additional propulsion requirement (see Fig. 3-10).

GROSS WEIGHT : 354,000 LBS.



PAYLOAD : 37,240 LBS.

STAGE TWO:

Thrust : 30,000 LBS (CONSTANT)
Propellants : LO_2/LH_2
Propellant Tanks :

	DESIGN CAPACITY	LOADING VARIATIONS
LO_2 :	12,830 LBS	12,830 – 8,330 LBS.
LH_2 :	64,150 LBS.	64,150 – 41,650 LBS.

Pump Fed

STAGE ONE:

Thrust : 150,000 LBS. (CONSTANT)
Propellants : LO_2/LH_2
Propellant Tanks :

	DESIGN CAPACITY	LOADING VARIATIONS
LO_2	39,250 LBS	35,130 – 39,250 LBS.
LH_2	196,250 LBS.	175,650 – 196,250 LBS.

Pump Fed

Figure 3-84. Nominal Two-Stage Vehicle for Mars Orbit Mission

Stage and Propulsion System Weights

Some vehicle considerations that affect the payload weight into the Mars orbit are the structure weight of the tanks, the engine system weight and other inert weights of the stage. These weights can be expressed in terms of a propellant fraction. A propellant fraction (0.915) for each stage has been assumed throughout most of the study. To investigate the influence of the chosen propellant fraction upon the payload weights, two other propellant fractions have been considered. The results are plotted in Fig. 3-85. The payloads are for a situation where both stages are assumed to have identical propellant fractions.

In Fig. 3-85 the first stage propellant fraction has been retained as 0.915 and the second stage propellant fraction results from various propellant combinations within the range of specific impulses considered for the second stage of the space vehicle. For a more precise calculation of the effect of inert weight, (i.e., structure weight, engine weight and other weights that are not directly payload) an analytical design study was performed and is presented in Section III. Using this information payloads are indicated in Fig. 3-85 and 3-86 for a mission begun on 30 November 1964. The results of the analysis indicate that the 0.915 propellant fraction selected as the nominal value for the studies of the Mars mission was only slightly too high. Thus, the selection of a nominal value of propellant fraction for this study is justified.

Specific Impulse

From the cryogenic propellant storage analysis, storage of liquid oxygen/liquid hydrogen is feasible, and desirable from a performance standpoint, for the Earth escape stage. Thus, for the parametric study the first

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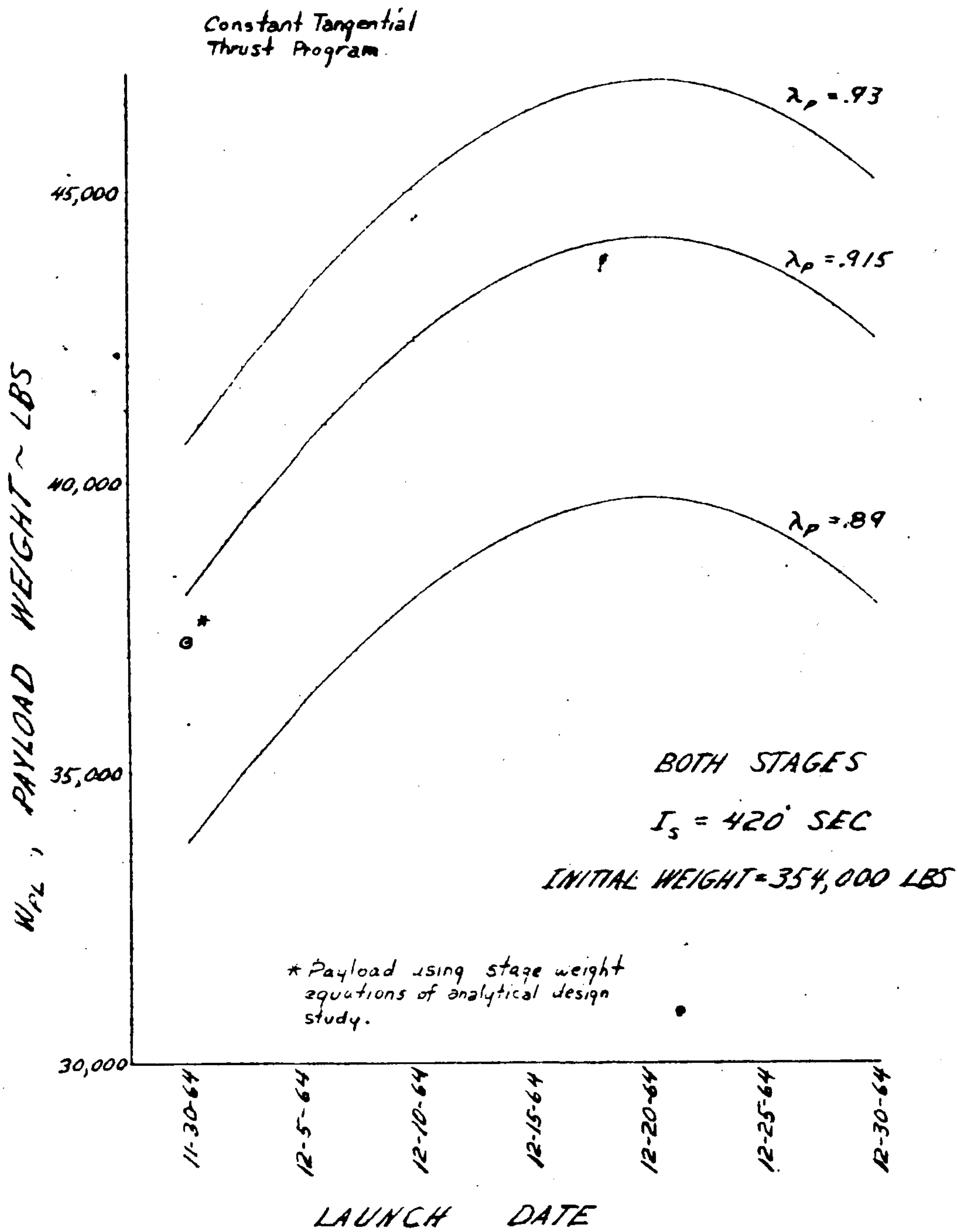

Figure 3-85. Effect of Propellant Fraction on Payload of a Mars Orbit Establishment Mission

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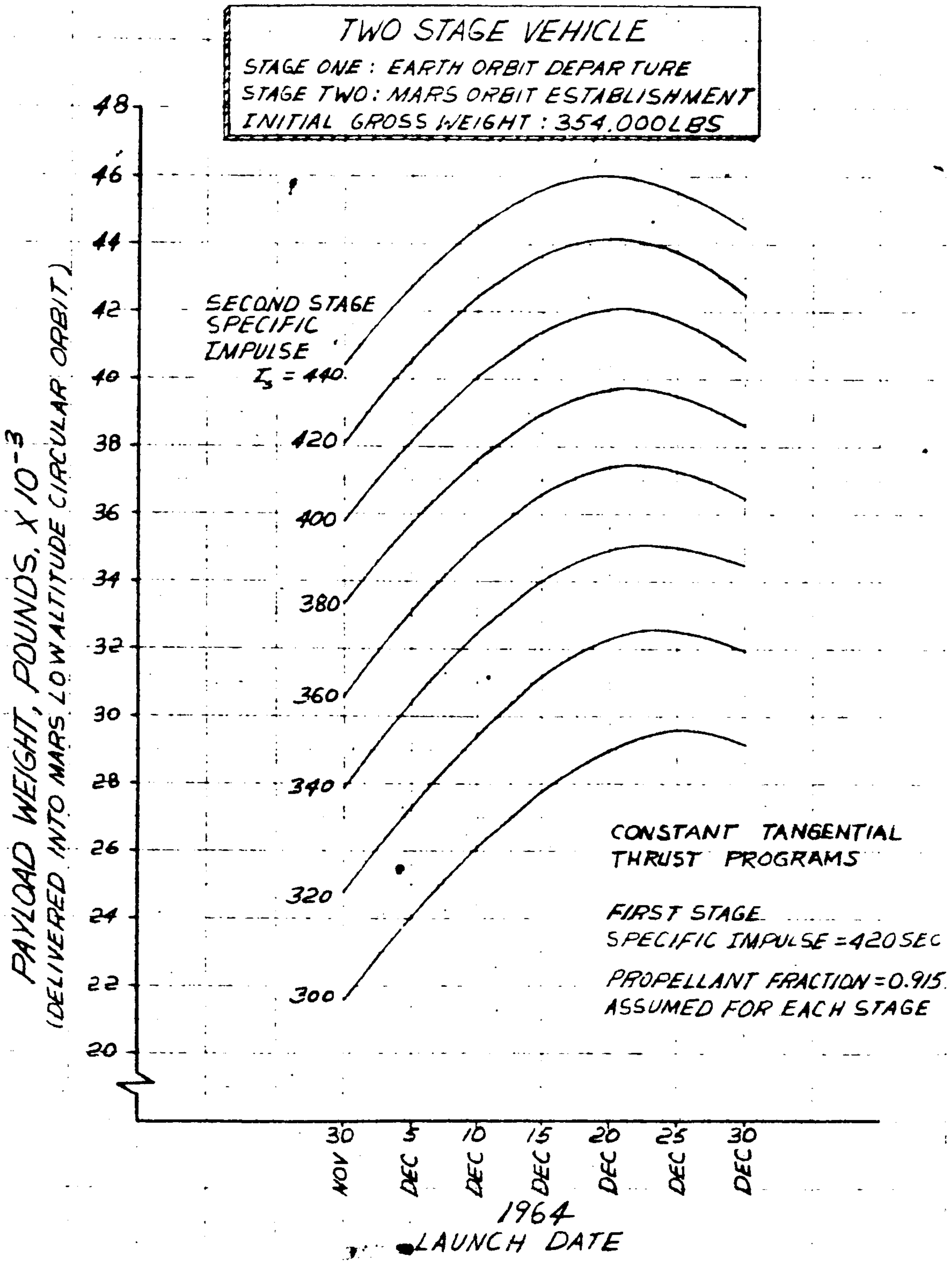

Figure 3-87. 200 Day Mars Transfer

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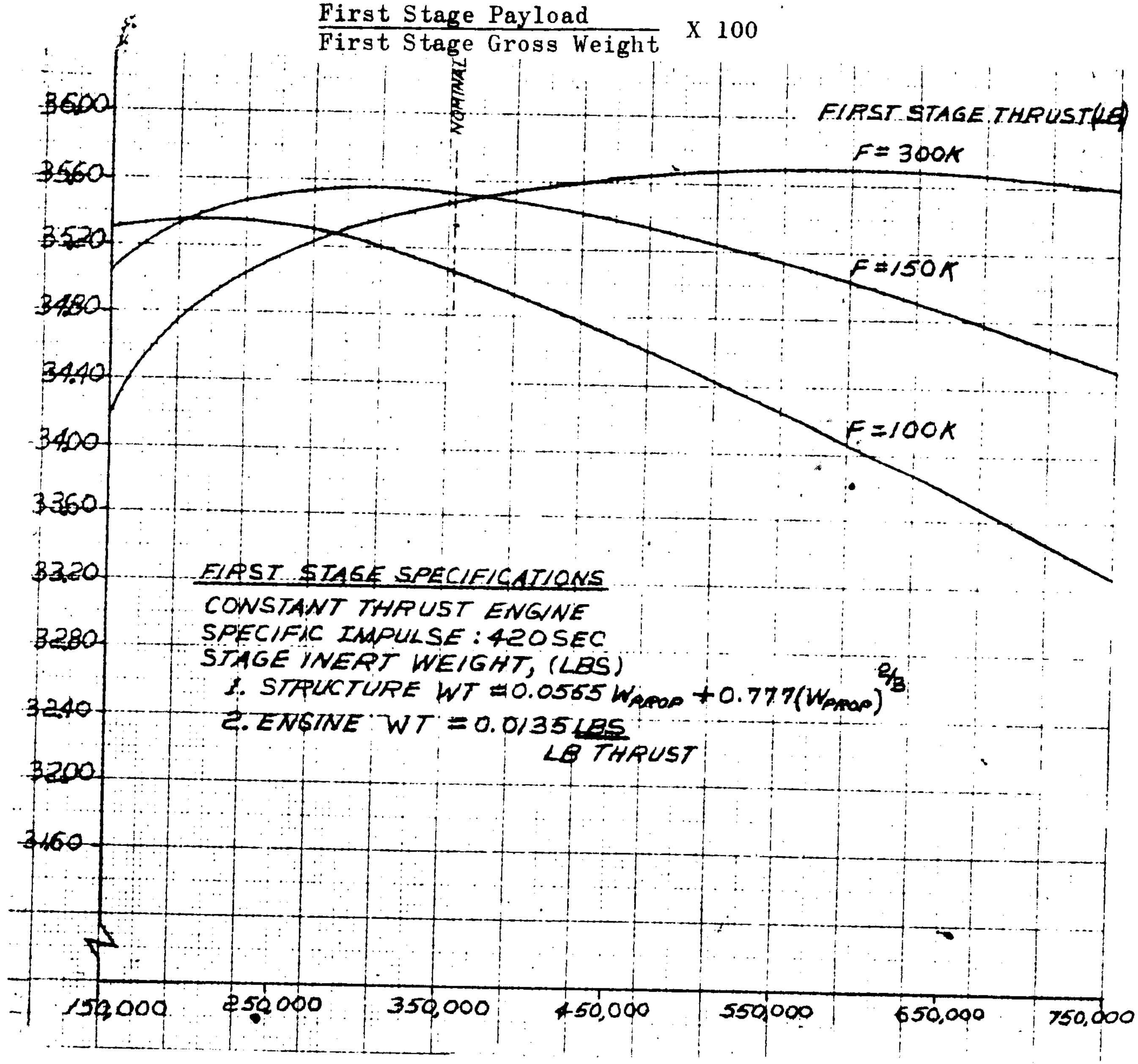

Figure 3-88. 200 Day Mars Transfer, Earth Orbit Departure on 30 November 1964

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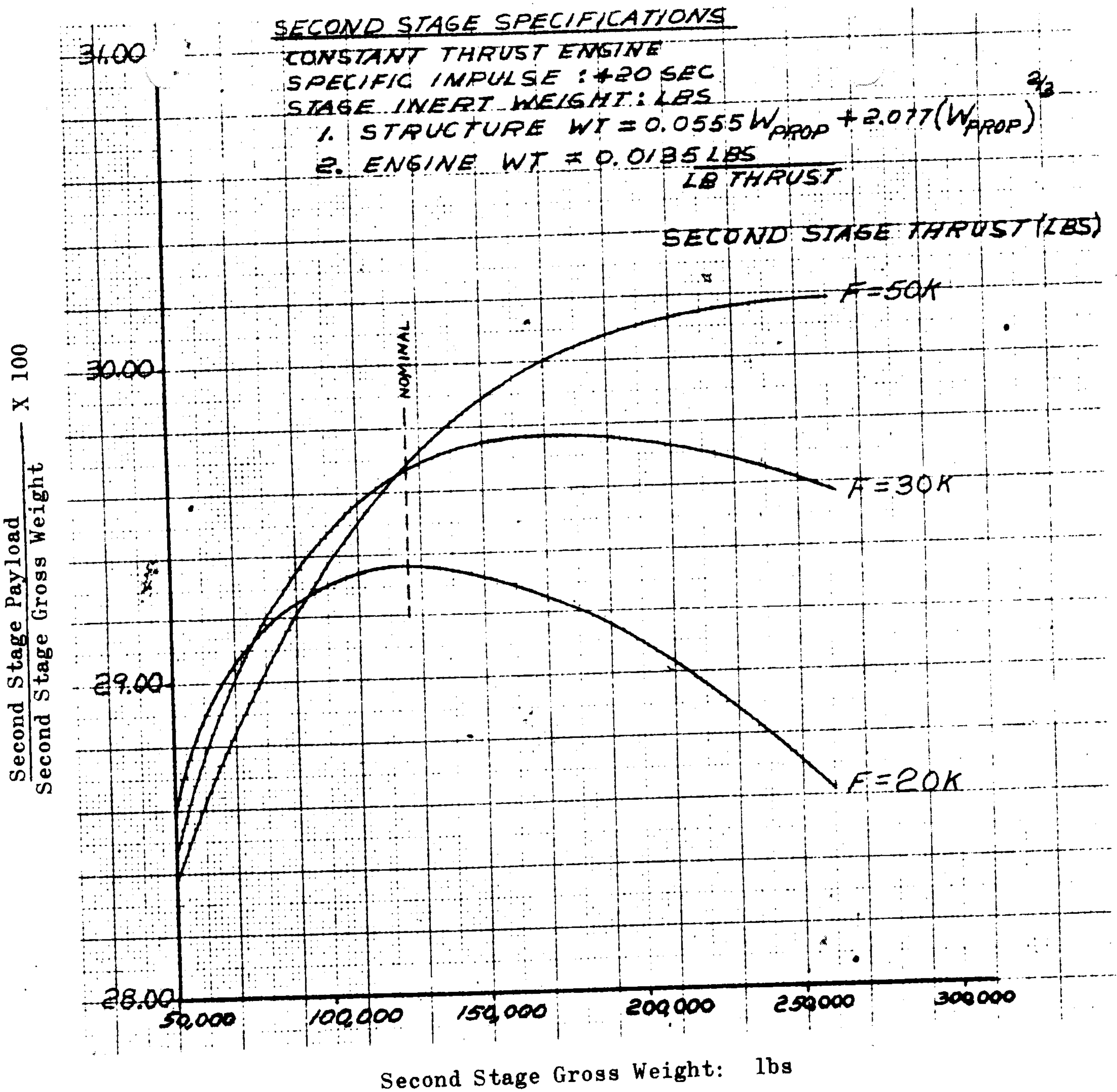

Figure 3-89. 200 Day Mars Transfer, Earth Orbit Departure on 30 November 1964

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The stage weights of the nominal two stage vehicle are identified to exemplify the effect of the thrust of the stage upon the stage payload that can be realized. For the first stage, a thrust level of 150,000 lb (for the nominal value of the 354,000 lb space vehicle) was recommended in Phase 1 of the study. If the initial gross weight of the vehicle increases much beyond the 354,000 lb nominal case, the velocity requirements increase and the percentage of the payload to gross weight begins to drop off. An increase of up to 650,000 lb initial gross weight shows a decrease in performance of less than 1 percent below that obtained using a higher thrust level of 300,000 lb. As the initial gross weight decreases, the inert weight assumes a more predominant role and the obtainable payloads/gross weight first shows a slight gain; then begins to decrease. This means that the 150,000-lb thrust is not exactly optimum for the vehicle, but is extremely close to optimum.

As previously mentioned, the second stage with its long transfer times has a strenuous propellant storage requirement. This requirement is such that the insulation weight required to maintain the cryogenic, liquid oxygen/liquid hydrogen propellant combination may decrease its payload capability so that a system using storable propellants would be more desirable. Preliminary consideration of this problem (Section III) indicates that for the nominal vehicle size or a larger vehicle, the liquid oxygen/liquid hydrogen combination would be desirable. For vehicles much smaller than the nominal, the storable propellant combination begins to be attractive.

For the second stage of the space vehicle a thrust level of 30,000 lb was recommended for the nominal case. Again, the percent payload to initial gross weight of the second stage shows peaks for each thrust level when plotted as a function of the initial gross weight of the second stage.

[redacted]

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The curve for a thrust level of 20,000 lb peaks at about the stage weight of the example used. However, as noted from the curves, an increase in performance is realized using the recommended 30,000-lb thrust level.

Thrust Magnitude

One of the propulsion system parameters that may vary for the retrothrust maneuver is the thrust magnitude. The propulsion system may not operate at the nominal design thrust level. Thus, it is of interest to determine any deleterious consequences upon the resulting orbit about Mars. The variation in the thrust level causes different effects depending upon the type of retrothrust maneuver termination employed. If the guidance and control system is programmed to stop the retrothrust maneuver after a selected period of time, the resultant orbit will vary accordingly (Fig. 3-90).

However, a stage designed to provide a given velocity increment or fixed total impulse (or to terminate upon depletion of stage propellant weight) would be recommended. The orbital deviations from this termination maneuver are presented in Fig. 3-91 . The major axis of the resulting orbit changes by less than 100 n mi if the propulsion system operates at a thrust deviation as large as ±1.5 percent of the nominal 30,000 lb. Although this investigation was conducted for a launch date of 30 November 1964, it is believed to be representative of the results that would be obtained for other launch dates during the 1964 period.

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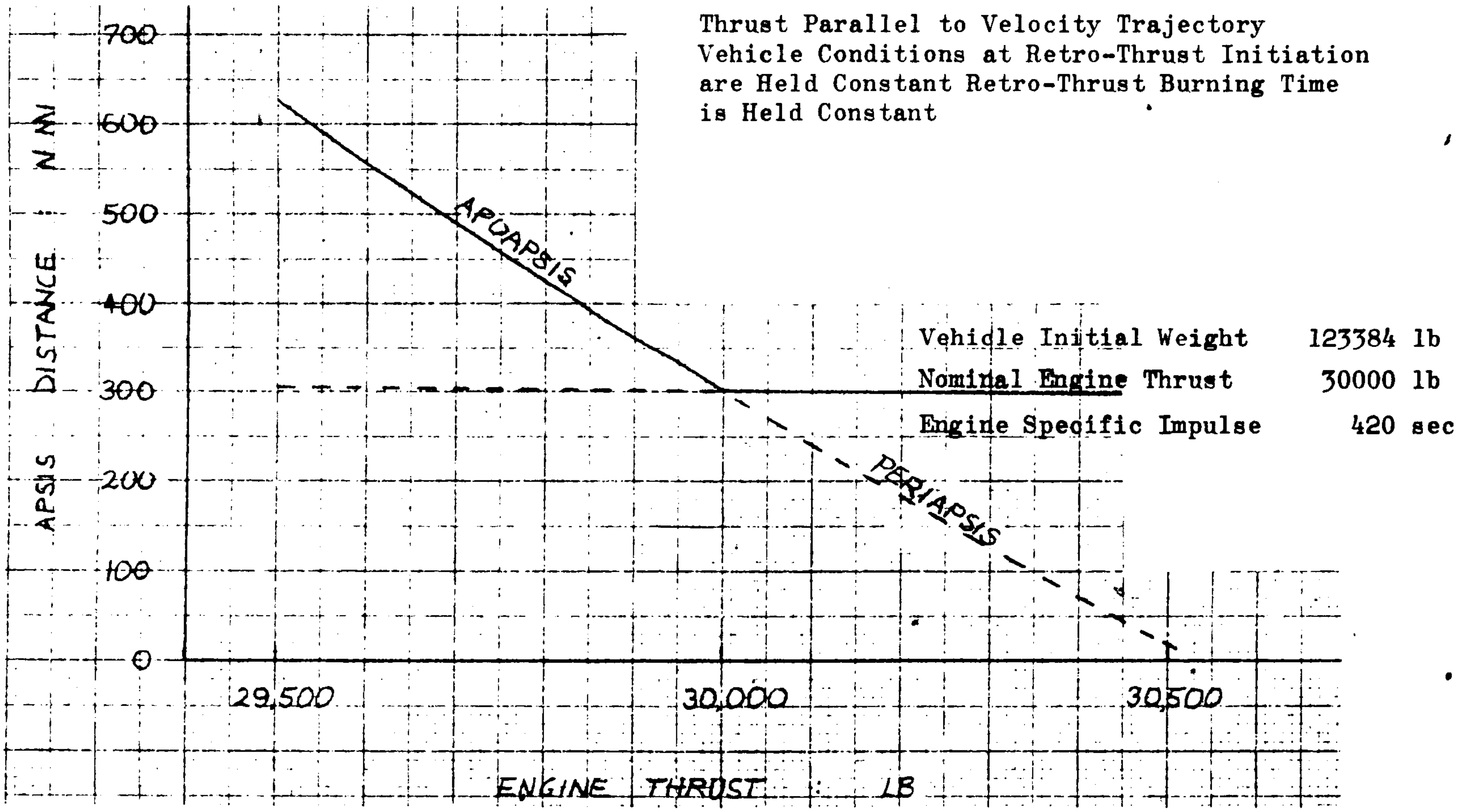

Figure 3-90. Mars Orbit Variation Due to Engine Trhsut Variation (Launch Date 30 November 1964)

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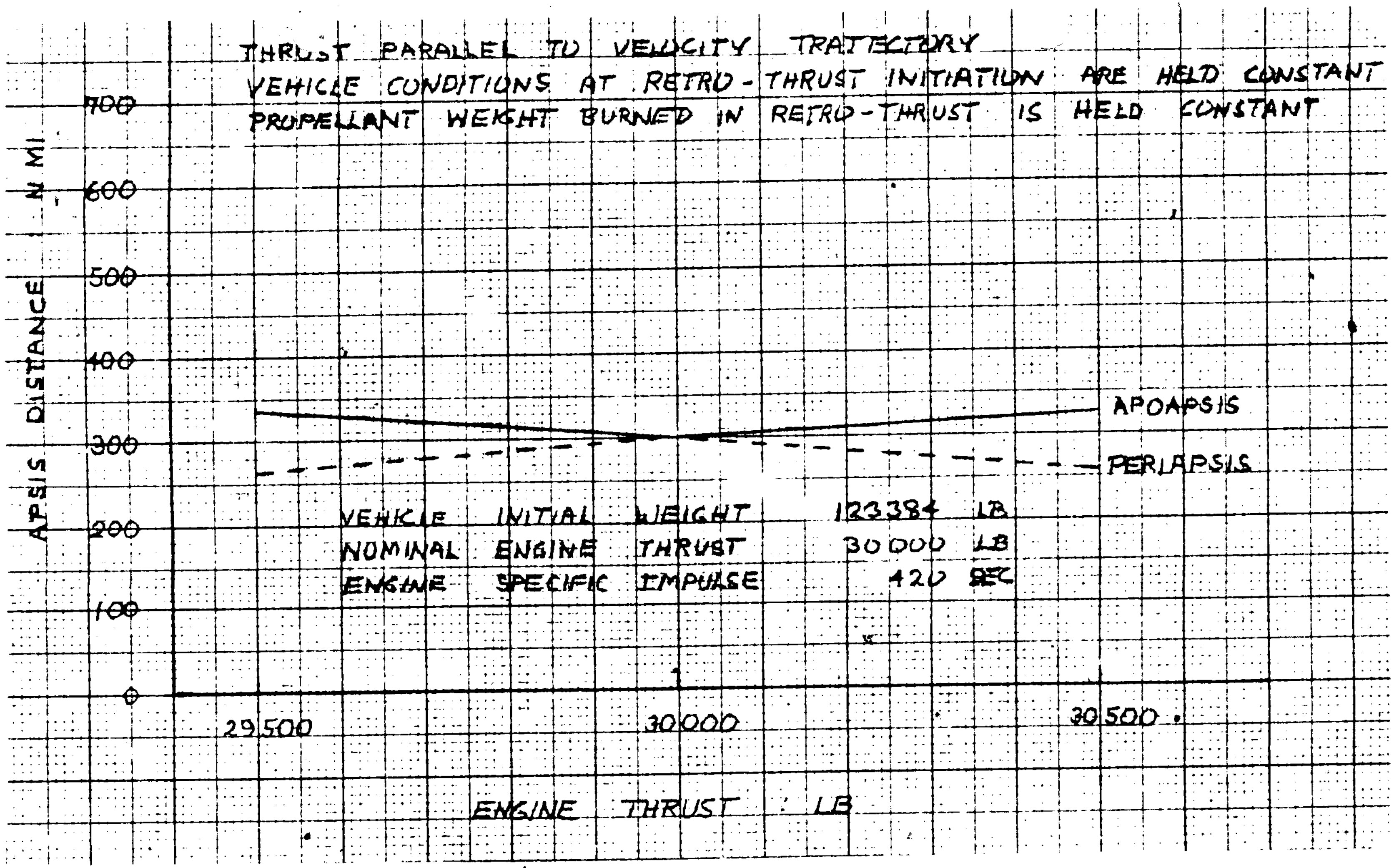

Figure 3-91. Mars Orbit Variation Due to Engine Thrust Variation
(Launch Date 30 November 1964)

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Transfer Time

In addition to the propulsion system parameters, the transfer time between the Earth and Mars plays an important role in the payload magnitude. As discussed previously a 200 day transfer has been recommended as the nominal. This transfer time does not yield the largest payload capability for the 1964 departure period (Fig. 3-92). Selection of 220 days would ameliorate the payload capability for this departure period. However, recall that the optimum transfer time varies with the year of departure, and is oscillatory about 200 days. Although the nominal has been recommended as 200 days, the transfer time could be varied each favorable departure year to gain payload weight.

The payloads presented in Fig. 3-92 are based upon trajectories using a constant retrothrust tangentially applied. A shift in the transfer time would require additional analyses to ascertain the thrust program and trajectory parameters for establishing the Mars orbit. The asymptotic approach distance and other parameters as presented in Fig. 3-78 for the 200 transfer time would have to be evaluated for each transfer time.

Mars Orbit Selection

Most of the orbit establishment studies have been based upon a 300 n mi circular orbit. Obviously, less payload can be placed in a 300 n mi circular orbit for a particular hyperbolic approach velocity than can be placed in an elliptic orbit with a perigee of 300 n mi.

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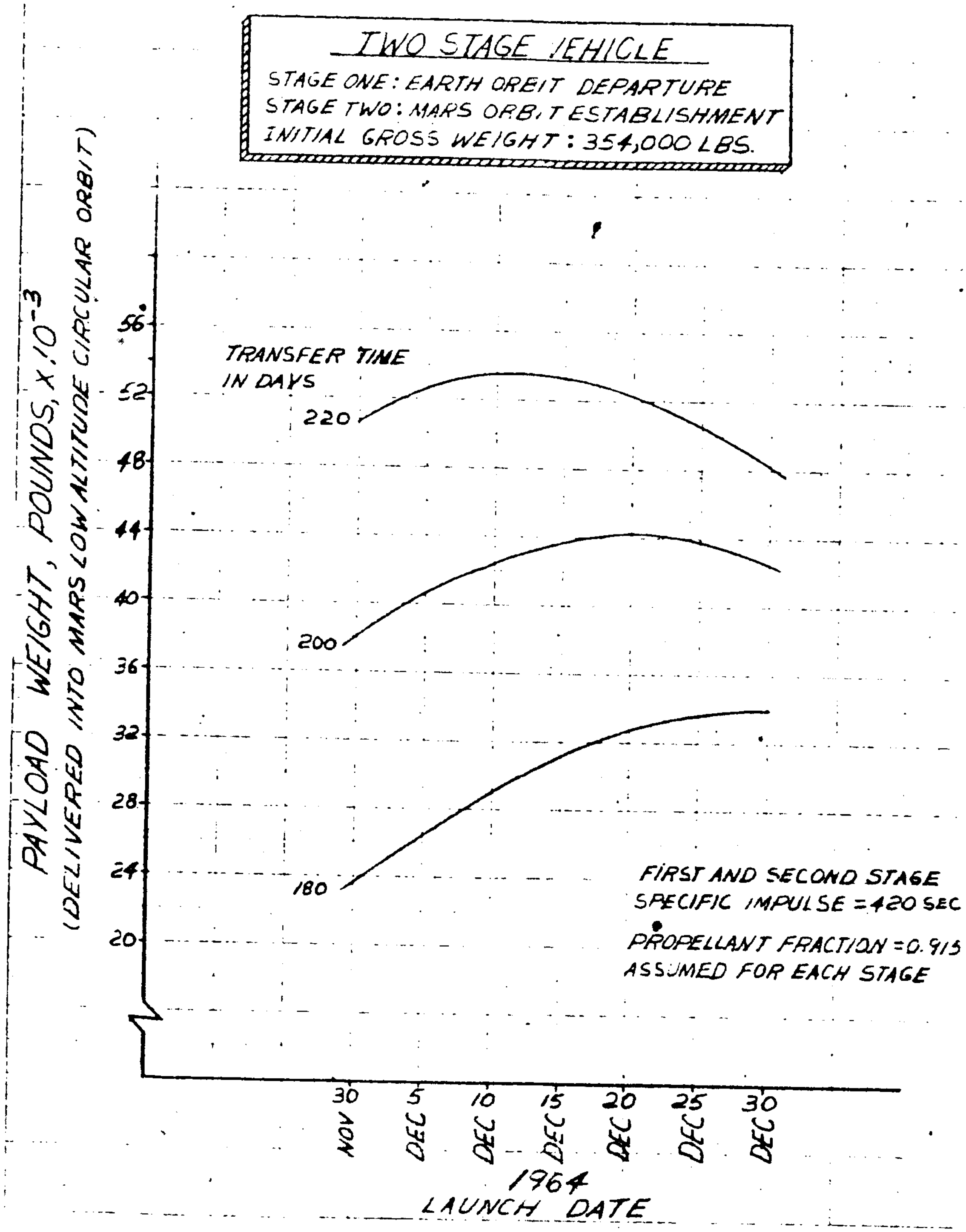

Figure 3-92. Mars Transfer Mission

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To determine just what effect the shape of the Mars orbit will have on the magnitude of the payload, a study was conducted for a representative launch date (30 November 1964) for the nominal vehicle. The orbit establishment maneuver used constant retrothrust applied tangential to the trajectory. The retrothrust maneuver was initiated at the altitude for establishing a 300 n mi circular orbit as presented in Fig. 3-78 .

The variable in this analysis was the second stage propellant weight. Off-loading of second-stage propellant from that required to establish a 300 n mi circular orbit decreases the burning time; thus, the resultant orbit deviates from the circular orbit. The resultant orbital elements are presented in Fig.

As much as 8000 lb payload can be gained by off-loading this identical amount of second-stage propellant without a significant change in the perigee altitude of the 300 n mi. The apogee altitude and the eccentricity of the resulting ellipse do change appreciably, as shown by the curves plotted in Fig. 3-93 . Depending upon the nature of the Mars orbit mission for an instrumented vehicle, significant gains in payload can be obtained in this manner by accepting an elliptical orbit instead of a circular orbit. This, of course, will depend entirely upon whether or not the apogee distances and eccentricities are compatible with the goals of mission.

A VEHICLE DESIGN FOR LAUNCHING DURING A 1-MONTH INTERVAL OF DATES FOR THE 1964 OPTIMUM LAUNCH PERIOD

The preceding plots of payload vs launch date assumed space vehicles (departing an Earth orbit) to be specifically tailored to the propellant and velocity requirements of each departure date. It is recommended

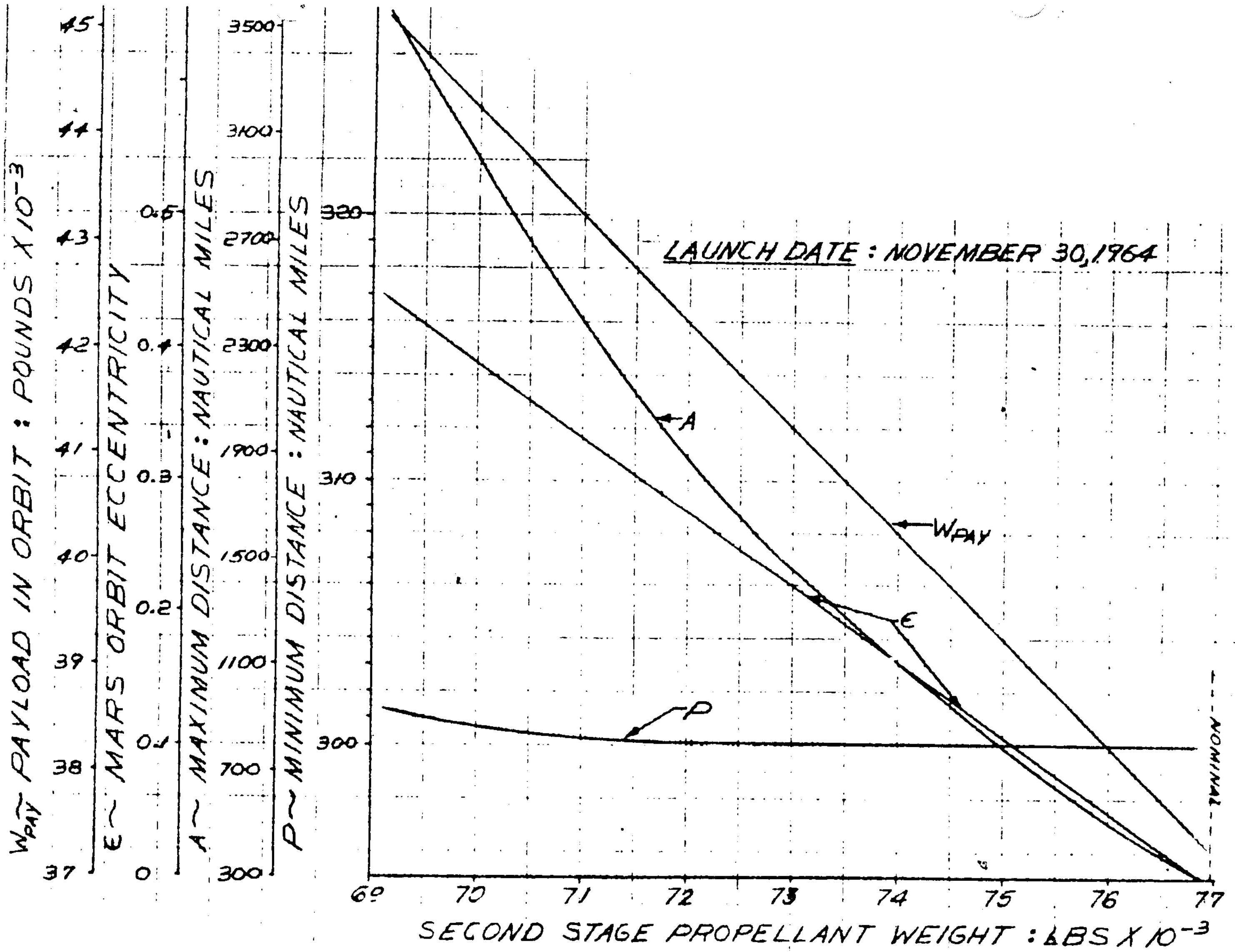

Figure 3-93. 200 Day Mars Transfer, Elliptical Path Description Around Mars and Payloads Resulting From Second Stage Propellant Loadings

that a vehicle be designed such that it has the capability of being launched any time during a 1-month interval of launch (Earth orbit departure) dates without extreme deleterious effects upon the payload capabilities of the vehicle. Actually, this means designing propellant tanks of each stage large enough to accommodate the maximum propellant volume required for any of the launch dates throughout the 1-month interval. During the 1964 period of launch dates, an interval of 30 November 1964 to 30 December 1964 has been selected because the launch date for maximum payload occurs during this interval.

The propellant tanks of the space vehicle are designed for the maximum volume expected for any launch date during this period. The maximum propellant requirements of the first stage occurs on 30 December 1964 (Fig. 3-55). The structure and tank size for the first stage will be designed for this propellant loading and its required residual propellants. The second stage of the space vehicle, the stage for establishing the Mars orbit has a maximum propellant requirement on the 30 November 1964 departure date (Fig. 3-58). Thus, the second stage tank size will be designed to accommodate the propellants required for this maximum value.

This design approach reduces the maximum payload for any given date by the difference in design tank/structure weight over the minimum tank/structure weight to satisfy the propellant requirements of that departure date. Thus a penalty is paid in the "theoretical" payload of a vehicle by designing the stages according to this method. (The word theoretical is introduced here to indicate that in subsequent paragraphs discussion will be presented to show that this so-called payload loss is meaningless when other factors of a 1-month launch period are evaluated.) Figure 3-94 shows the loss in these theoretical payloads as a function of the launch date.

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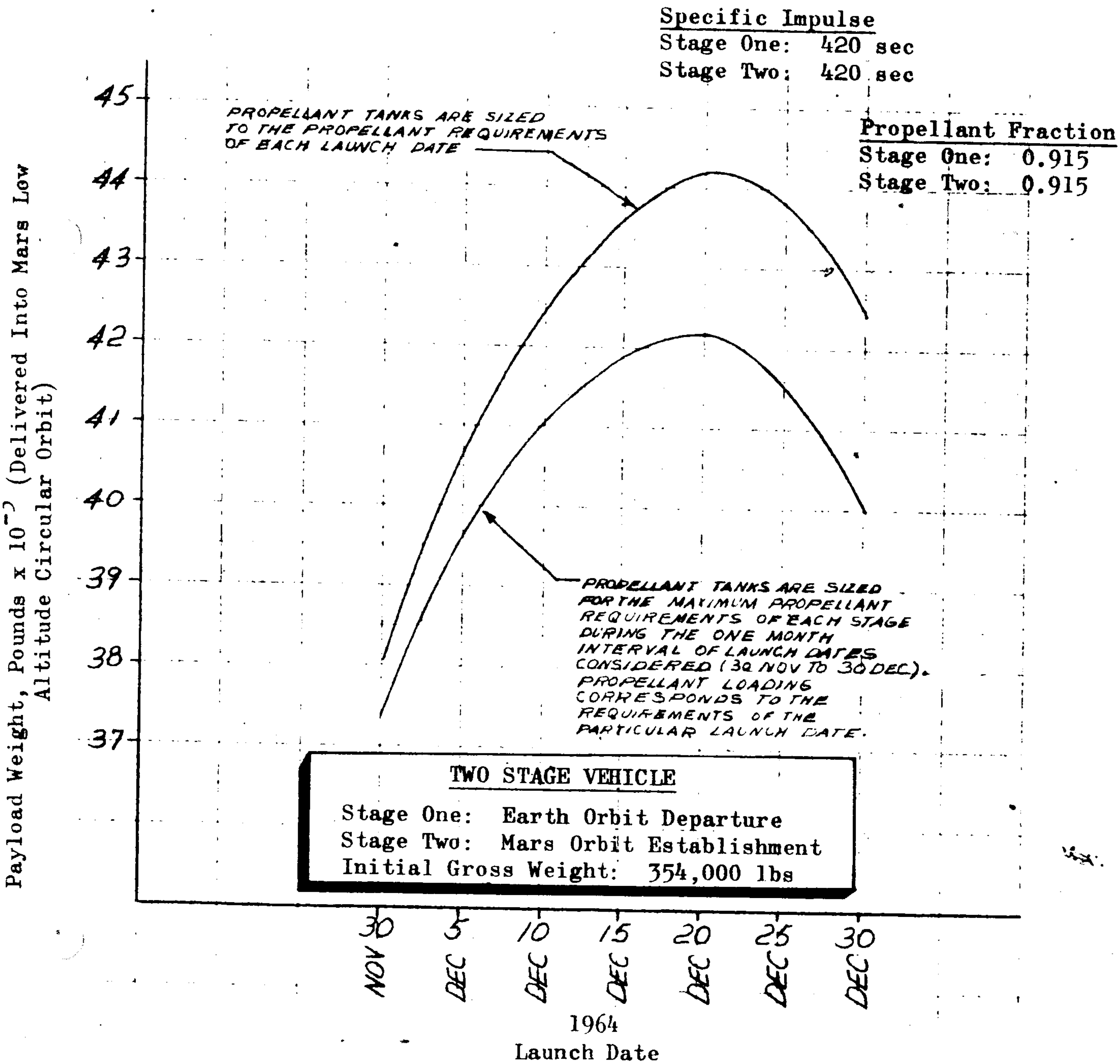

Figure 3-94. 200 Day Mars Transfer (Two Stage Vehicle)

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An interesting aspect arises when a space vehicle is recommended as applicable to any departure date from an Earth orbit within a specified interval of dates. Specifically the problem is this; how can the propellant loadings of each stage be changed to the requirements of a particular launch date once the space vehicle is in an Earth orbit awaiting a departure. The easiest solution to this problem is to suggest that the vehicle be placed into the Earth orbit immediately prior to departure. Thus, the proper propellant loading could be accomplished on the ground. However, the boost-to-orbit trajectory, consistent with the recommended proper inclination of the geocentric parking orbit, may severely reduce the 354,000 lb initial gross weight of the space vehicle.

A possible alternative that warrants detailed evaluation (beyond the scope of this study) would place the vehicle in a 28.5 deg orbital plane prior to Earth orbit departure. This 28.5 deg inclination plane could be changed by some plane change propulsion maneuver to achieve a desired geocentric orbit plane inclination during the time interval between Earth launch and Earth orbit departure. If the geocentric plane change maneuver requires slightly more or less time than allowed between Earth launch and Earth orbit departure, the propellant requirements of each stage change accordingly.

As indicated in Fig. 3-95 the shape of the curve of the total propellant weight consumed by the combined two stages for the mission is practically flat. (The second stage propellant weights are for a Mars orbit establishment maneuver as described in Fig. 3-78.) This suggests the possibility of loading a fixed amount of propellant on the space vehicle prior to

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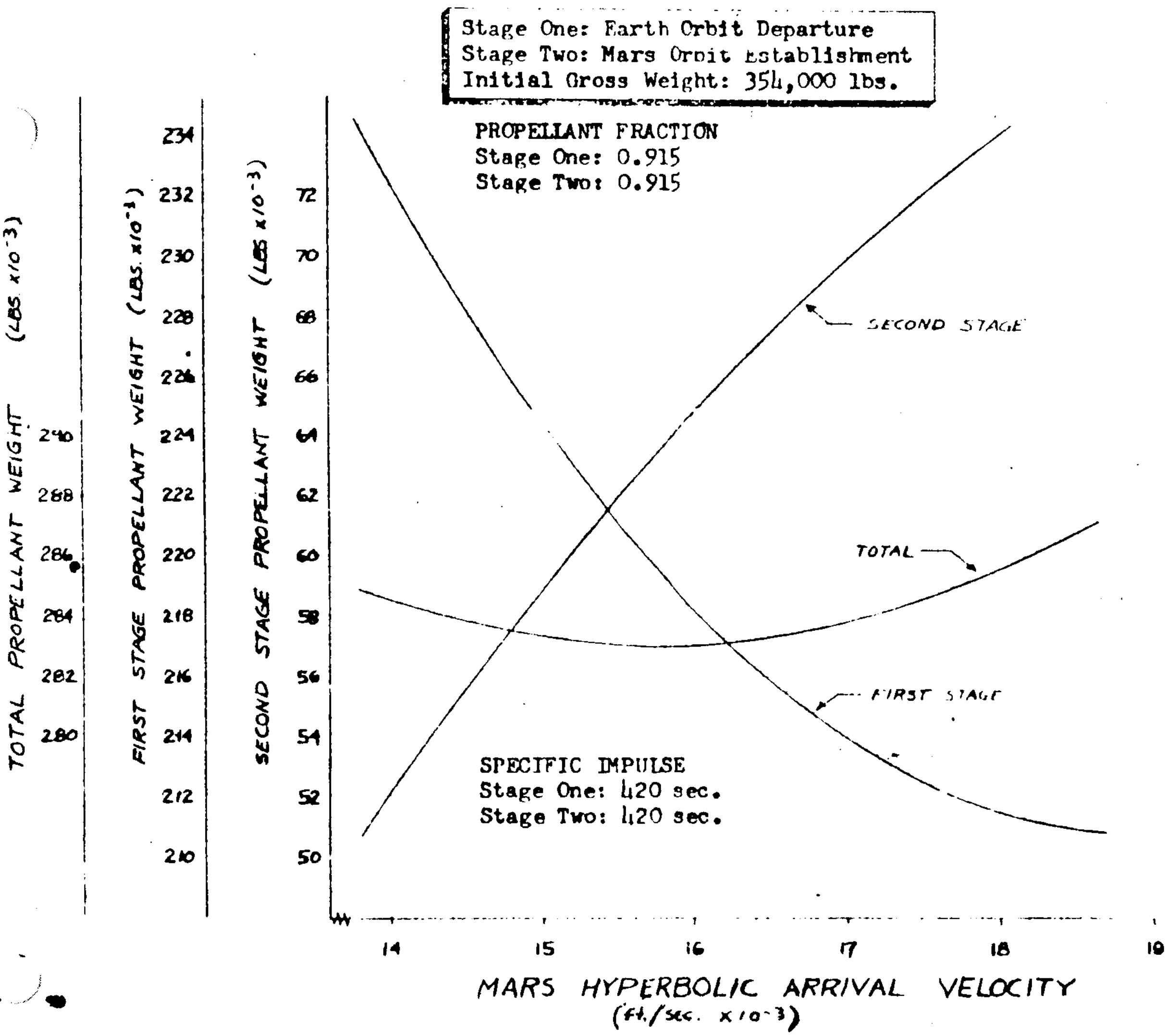

Figure 3-95. 200 Day Mars Transfer, 1964 Launch Period. Constant Tangential Thrust Programs

[redacted]

orbit establishment (utilizing the recommended tank design) adequate to perform the mission for any departure date of the interval. The tanks of the stages might be interconnected to permit shifting of propellants after the vehicle is in a geocentric orbit. Regardless of the particular departure date the required propellant loadings could be secured for each stage.

According to Fig. 3-94, a space vehicle will have a varying payload capability which is a function of the launch date. (The payloads are computed using trajectories as described in Fig. 3-78.) This payload capability variation is a direct function of the energy requirements. The payloads presented in the figure must be recognized as theoretical payloads only, if one vehicle is designed to be launched on any date during the period. To be realistic about payload weight, a nominal value must be selected for the interval of launch dates which necessarily could be the minimum value during the period. Otherwise, continuous changing of the payload package would be required when the launch date changes. It is impractical to vary the payload weight of a particular vehicle during the month interval.

Realistic payloads must be accepted as the lower limit if a particular vehicle is to be capable of geocentric orbit departure any day during the period. From Fig. 3-94 the payload would be selected as approximately 37,240 lb. Any difference between the theoretical payload as shown by the curves and an actual payload could be residual propellants in the second stage. This amount of residual propellant could be used for maneuvering the vehicle once it has established the 300 n mi circular orbit.

It is of prime interest to consider designing a vehicle that could be used for more than one particular mission or for one particular launch date. The design of this nominal vehicle has been shown to be capable of launch during a 1-month interval of launch dates during the 1964 optimum period for a Mars transfer. A similar optimum time occurs for launching a Mars transfer vehicle during a period encompassing December 1966 and January 1967. The nominal vehicle as designed for the 1964 launch period has been analyzed for its adaptability to the 1967 departure period. The identical vehicle (an initial gross weight of 354,000 lb) with both stages designed as recommended for the maximum propellant loadings of the 1964 launch period has been used.

During the 1967 interval the energy requirements are somewhat less than the 1964 interval. Therefore Fig. 3-96 shows a gain in theoretical payload for a Mars mission. A similar analysis could be conducted for o other optimum departure periods. Other optimum periods have not been considered for inclusion in this report. Launch date in 1969 could also be considered. However, for this nominal vehicle it is believed that only the payload curve would shift, assuming all maneuvers were the same.

using one stage over the other for the orbit establishment are a result of a more optimum distribution of ideal velocity requirement between the two stages involved.

LUNAR LANDING AND RETURN MISSION

The study of any propulsion system must necessarily involve an analysis of the mission which the system is to accomplish. The large number of maneuver alternatives which are possible to accomplish a space mission require that these maneuvers be investigated in detail to fully exploit the potentialities of the propulsion system. Figures 3-101, 3-103, 3-112, and 3-114 show the wide variety of maneuvers considered step by step for the lunar mission. The shaded areas on the figures represent the maneuvers and thrust profiles selected. The following sections contain descriptions of these selections, the reasons for their choice, and brief analyses to indicate the effects of system errors on the maneuver termination conditions.

The three major divisions are Earth-Moon transfer, lunar landing, and Moon-Earth transfer. Lunar landing embraces a particularly wide variety of alternatives which very strongly influence the propulsion system design parameters. This section has been subdivided into two areas so that actually two lunar missions are presented; the landing and return of later manned vehicles.

Considerable use has been made of analyses performed in Phase 1, but further trajectory simulations and analyses (particularly for the landing maneuver) have been added to provide more complete coverage of the possible maneuvers.

[redacted]

INITIAL CONDITIONS

The starting point considered for the lunar mission was a 300 n mi orbit around the Earth. A description of the trajectory assumed to have placed the space vehicle in this orbit is given in a previous section. The use of a parking orbit, although slightly more costly in propellant expenditure than the direct transfer, greatly alleviates the restrictions on launch time and position to accomplish a given lunar transfer trajectory. These restrictions are discussed at length in Ref. 1b which describes the limits of the direct transfer trajectory.

An altitude of 300 n mi was selected for the parking orbit as a plausible compromise between increased propellant requirements and Van Allen belt radiation hazards in high altitude orbits, and orbital perturbation and decay in low orbits. As the orbit height increases more propellant is required to place the vehicle in orbit. In fact, the very high altitude orbits require more propellant than the escape mission. On the other hand, reports such as Ref. 17 indicate short lifetimes for the lower altitude orbits. The height selected for the parking orbit is higher than that required by the reference if only a few passes of the vehicle are necessary to obtain the correct alignment prior to transfer. However, the extra propellant requirement for a 300 n mi (compared to a 100 n mi) orbit is small and the orbital buildup and long stay capabilities are thus ensured.

[redacted]

Figure 3-101 shows the possibility of including an orbital plane change at this point. Although such a maneuver would reduce launch time restrictions it involves, generally, a rather high propellant consumption (velocity increments in the order thousands of fps) and was therefore not included.

TRANSFER TRAJECTORY

A 2.6 day transfer trajectory was selected based on considerations of propulsion requirements and crew requirements for manned missions. Increased trip time is beneficial to the gross payload capabilities of the vehicle because the lower velocity requirements result in less propellant consumption. This is shown in Fig. 3-102 where the upper curve represents the above mentioned gross payload. Further consideration must be given, however, to the effects of trip time on other parameters. For example, the life support system weights discussed in a previous section increase linearly with trip time for durations encountered in typical lunar missions. This weight is indicated as the difference between the upper and middle curves of Fig. 3-102 and is seen to have a small effect on usable payload for the lunar missions.

A far more significant factor causing reduction of the net payload is the radiation shield requirement for manned and other radiation sensitive payloads. The lower dashed curve shown indicates the drastic reduction in useful payload because of the use of shielding. The curve is dashed because a constant shielding weight was assumed and because of the relatively uncertain requirements for shielding (p 2-330).

MANEUVER COMBINATIONS FOR EARTH-MOON TRANSFER

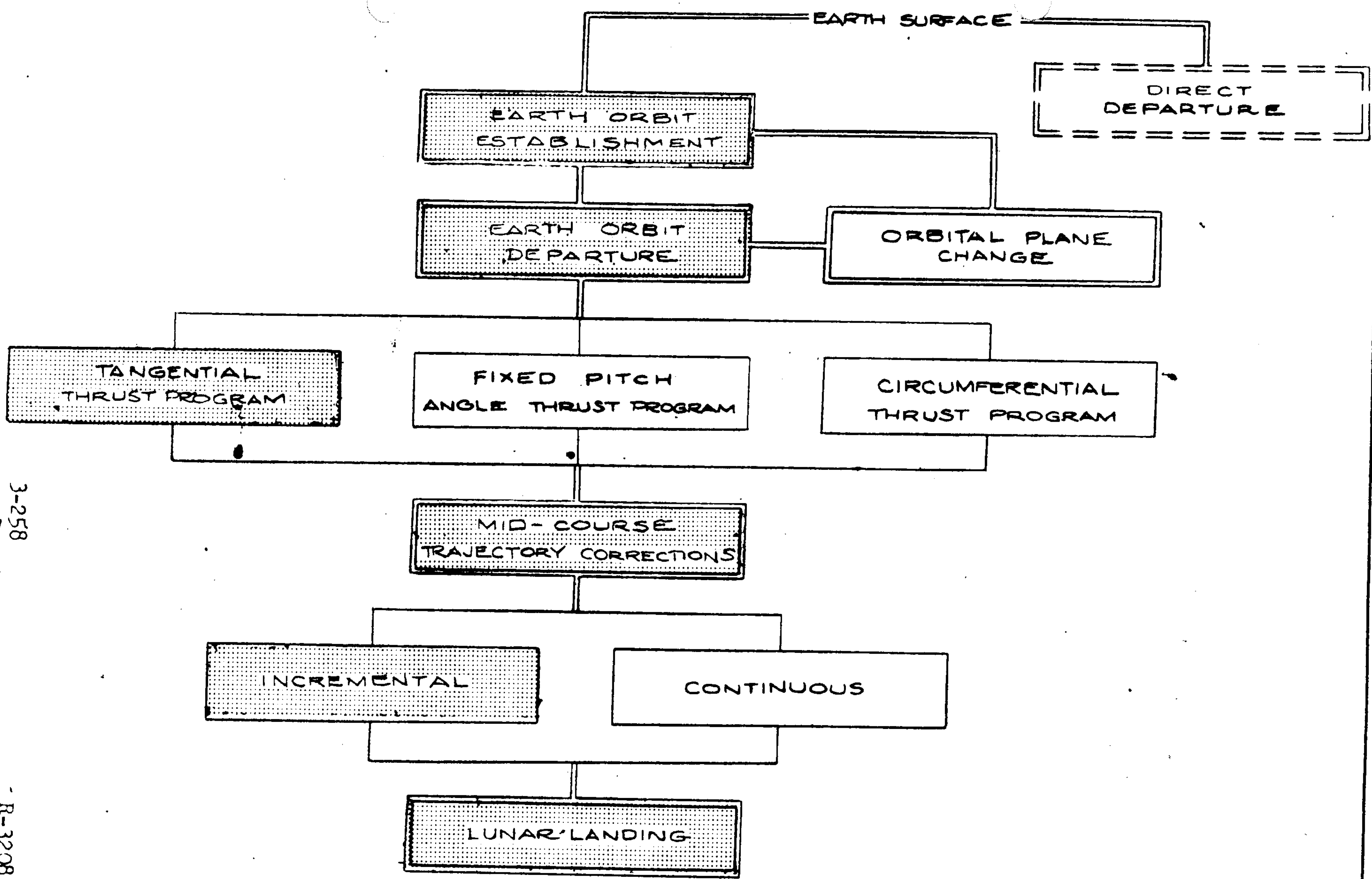

Figure 3-101. Maneuver Combinations for Earth-Moon Transfer

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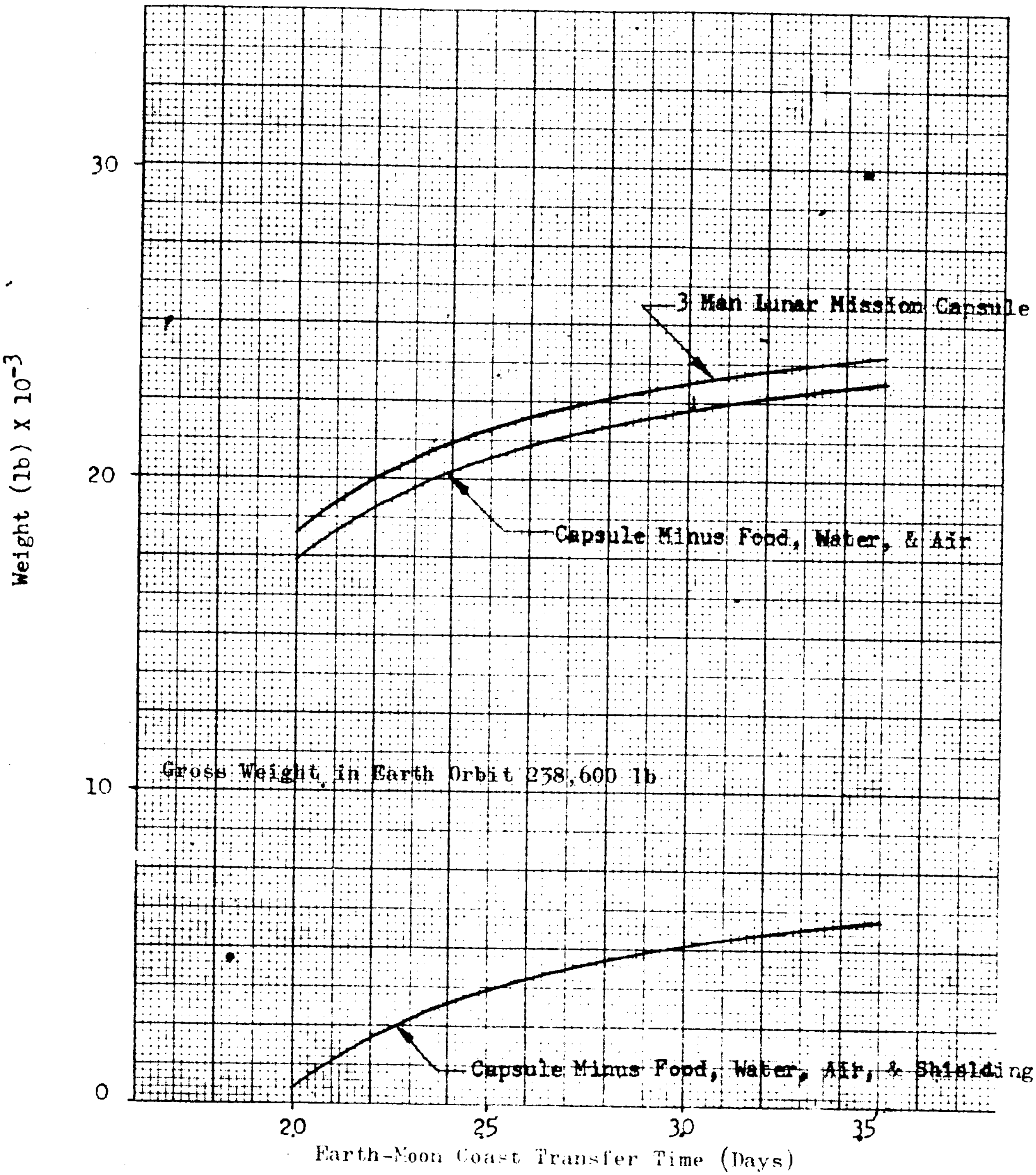

Figure 3-102. Capsule Weight Returned to Earth From Lunar Landing Mission

[redacted]

The shielding weight from which this curve was derived was based on an average trip time and is presented only to show the magnitude of the payload loss rather than any time dependence. In actuality, more shielding may be required as the trip time increased. First, the confidence level of solar flare predictability decreases as the duration to be predicted increases. Second, the probability of encountering a larger flare during flight increases as the flight time is extended. Thus, if the shielding thickness were based on an arbitrary probability (e.g., 99 percent) of receiving less than a specified dose of radiation during the trip, it is clear that the shield weight would be a function of trip time. Although more research must be accomplished before this function can be determined, the result of the increased shielding would be a peak in the next payload vs trip time curve. The 2.6 days used throughout this analysis is, therefore, not to be considered as a rigorous recommendation but rather a probable value of trip time.

A tangential thrust program was selected, based on simulated trajectory analysis, as the most economical with respect to propellant consumption.

The complete optimization of a midcourse correction program is a problem area concerned more with guidance equipment than with propulsion systems. Thus the investigation was centered around a literature study and the conclusions drawn are based primarily on the data and information contained in Ref. 3 , 4 , 5 , 6 , and 7 . These references indicate propulsion system velocity increments in the range of 25 to 200 fps. Both optical and radar sensing systems were considered, and single, dual, and triple corrections were evaluated. Errors in velocity and position of the vehicle as it approaches the moon as low as 3 fps and 1 n mi respectively are claimed. The small magnitude of even the most pessimistic velocity requirement indicated that further study of this phase was not necessary in this effort. Therefore, the relatively high value of 150 fps was assumed for the outbound midcourse correction and should be adequate for most guidance system requirements. The magnitude of the return midcourse correction was also assumed to be 150 fps.

A continuously applied correction using a low-thrust engine is possible, however the literature on the subject indicates that the demands of the guidance system have not yet been formulated. Furthermore, the subsequent propulsion system analysis indicated that the incremental propulsion system would also be applicable to other phases of the mission and was therefore used for midcourse propulsion.

LUNAR LANDING

Direct Landing

The maneuvers considered for landing are shown in Fig. 3-103. The shaded areas represent the recommended method of accomplishing the direct landing.

The early lunar missions may take the form of unmanned trajectories to serve as practice shots to test the early models of the propulsion, airframe, guidance, and other systems and also to accomplish scientific and engineering experiments on the lunar surface. The required landing CEP for these missions would therefore be quite broad (Ref. 19) since only a general obstacle-free area would probably be targeted and the restriction of landing areas available to this maneuver would be of little significance. To minimize guidance demands for these early

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MANEUVER COMBINATIONS FOR LUNAR LANDING

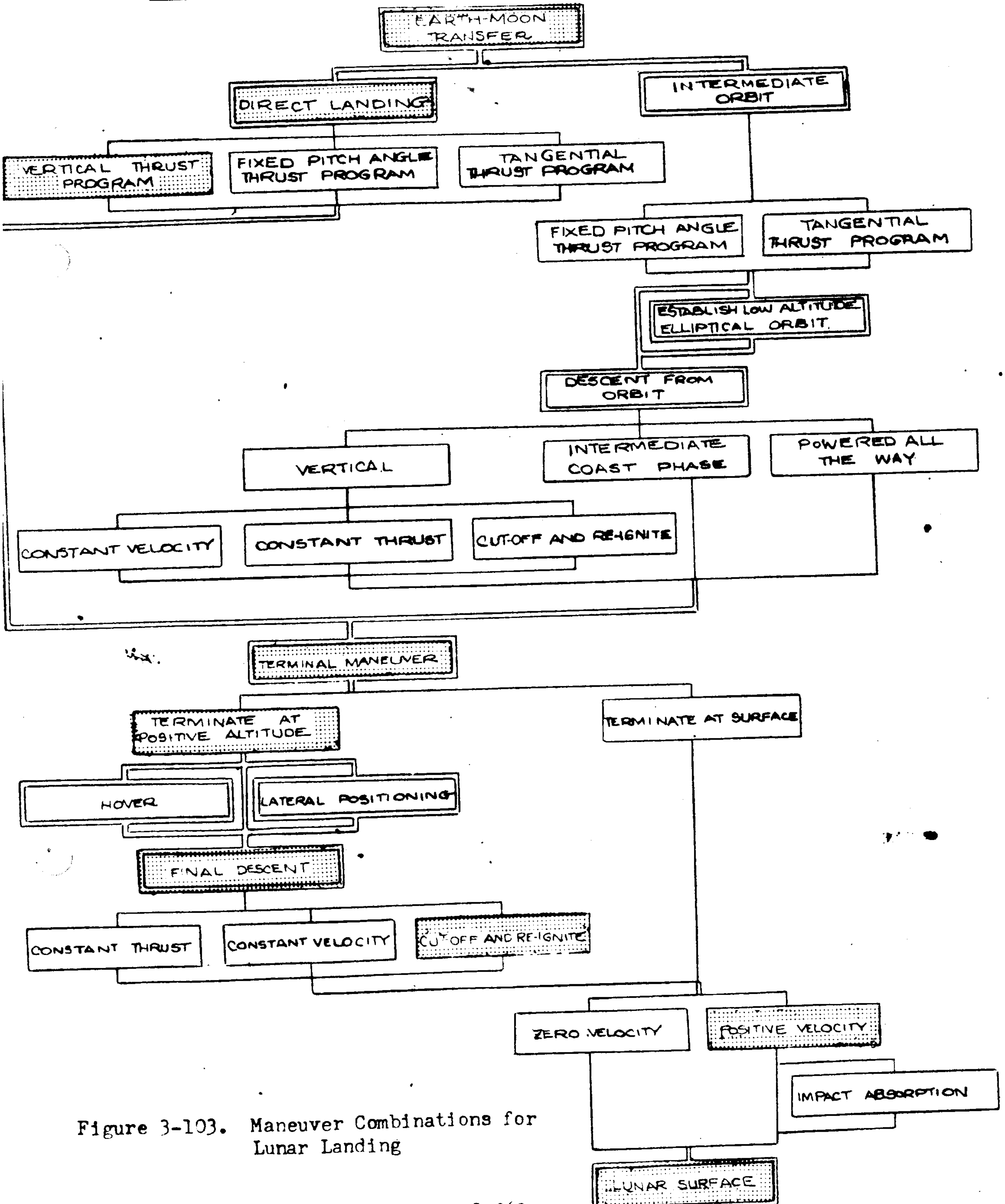

Figure 3-103. Maneuver Combinations for Lunar Landing

[redacted]

mission, a simple vertical descent landing following the sequence shown in Fig. 3-103 is recommended. This would require that the guidance system merely establish the local vertical, and measure the altitude to determine the time of thrust initiation. Reference 23 concludes that at the altitude where retro-thrust would be initiated, 1 percent accuracy in altitude measurement is possible if an optical horizon seeker were used for this purpose as well as for establishment of the local vertical axis. However, this accuracy would be improved by a factor of 10 by use of radar. Therefore, the system was assumed to use an optical horizon seeker to establish the vertical, and a radar sensor to determine altitude and descent rate. A constant-thrust restartable engine having a specific impulse of 420 sec was assumed for the trajectory study.

The analysis was begun by determining the altitude above the moon at which retro-thrust must be initiated (based on a 2.6 day transfer) to result in zero velocity at the lunar surface.

To simulate this maneuver the vehicle was flown vertically from the lunar surface, with the mass increasing with time. Figure 3-104 shows the resulting firing altitude as a function of the initial thrust to (lunar) weight ratio.

To ensure safety in the landing maneuver the possibility of errors in the velocity and altitude measurements was considered. Figure 3-105 shows the altitude at which the vehicle will reach zero velocity if the firing altitude and velocity were 0.1 percent high and low, respectively. Including the possibility of this error in the design the vehicle is then targeted for this altitude, the result being that the vehicle will reach zero velocity at an altitude between zero and twice that of Fig. 3-105 for combinations of errors in velocity and altitude up to 0.1 percent.

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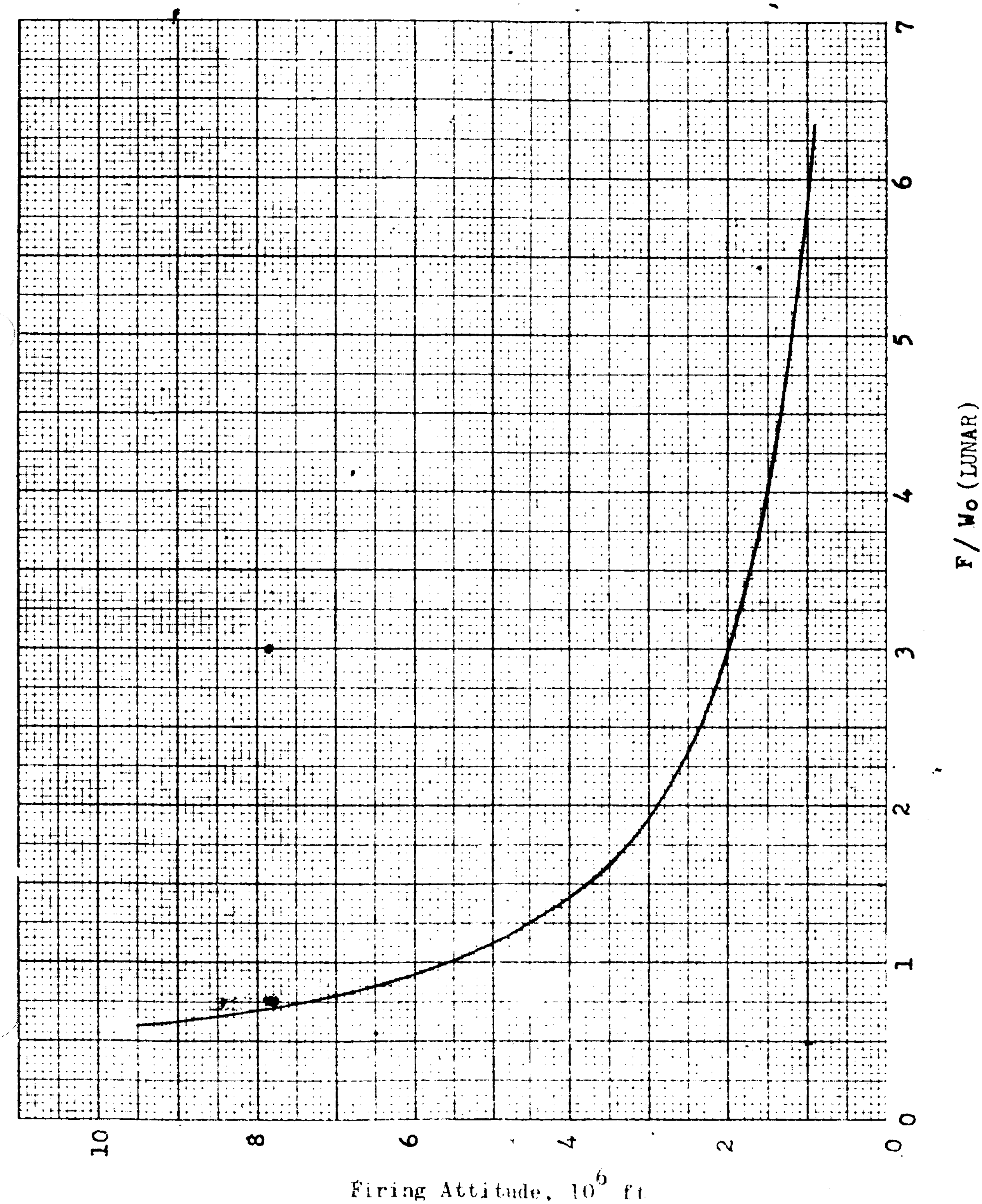

Figure 3-104. Initial Firing Altitude vs Initial Lunar Thrust to Weight Ratio

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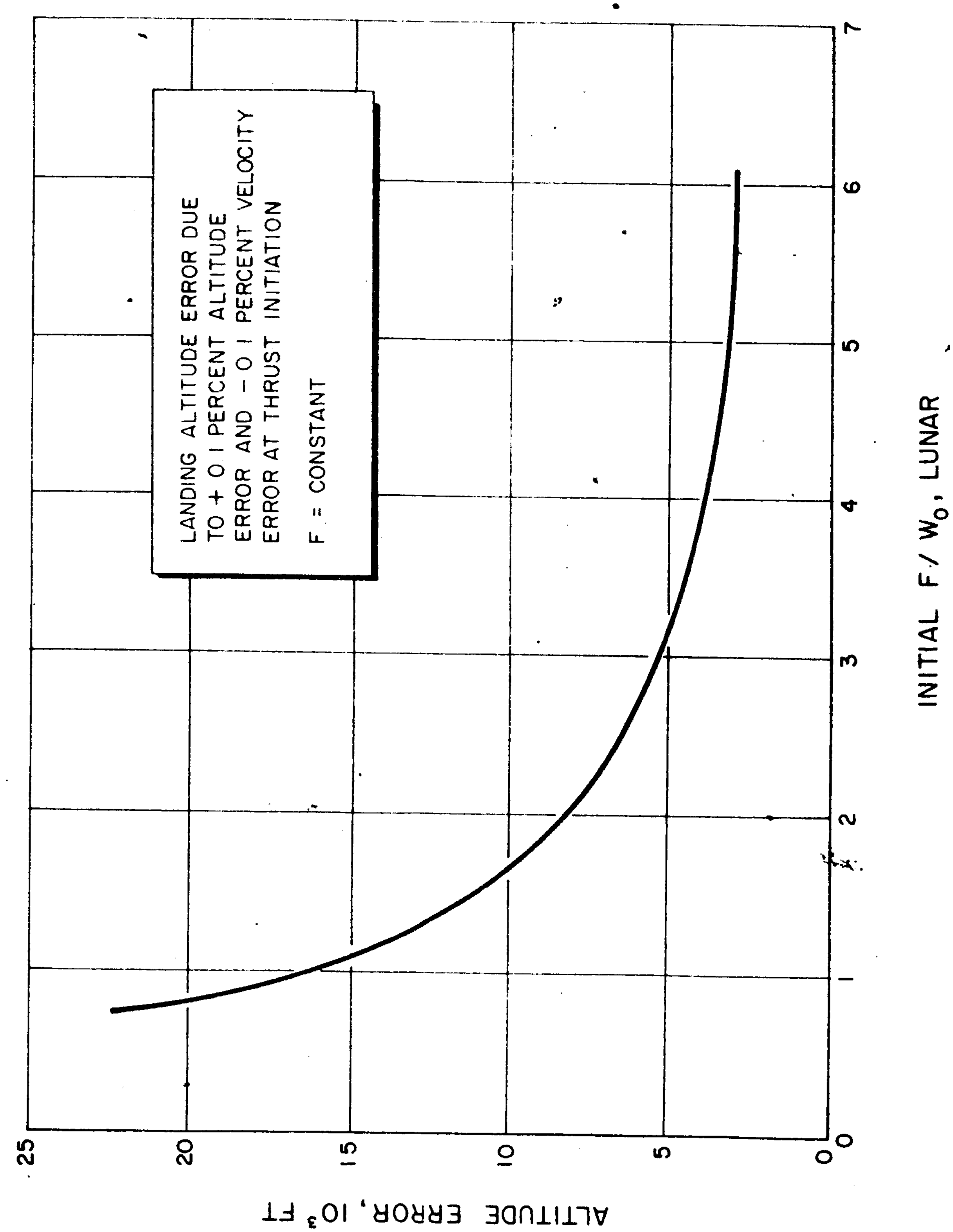

Figure 3-105. Lunar Landing From 2.6 Day Transfer

The magnitude of the burnout altitude resulting from measurement error at firing indicates that further propulsive maneuvers will be necessary. One possibility is that the vehicle free-falls to some altitude where the engines are restarted; another is that the engines are throttled and the vehicle falls at a small constant descent velocity; a final alternative is to reduce the thrust to some low level (F/W determined by first phase burnout altitude) and maintain this thrust level such that velocity would buildup and then decay to zero at impact. Figure 3-106 shows the propellant requirements of this final descent for the three maneuvers discussed above. The consumption is expressed on this graph in terms of the ratio of the propellant burned during the final letdown to the gross weight at the beginning of this period (or, equivalently, at the end of the first burning phase). This figure indicates the low propellant requirements of the cutoff and reignite maneuver compared to the others. This advantage coupled with the absence of throttling requirements led to the selection of this maneuver for the final descent phase. The principal disadvantage of this maneuver, namely the restart capability requirement, necessitates that the engine have a high starting reliability especially if the maneuver were used as part of a manned mission. (A discussion of a trajectory more suitable to landing manned payloads will be found in the next section.) The following discussion presents some aspects of the cutoff and reignite final descent maneuver.

Assuming that the worst conditions prevail, the vehicle will be at an altitude of twice that shown in Fig.3-105 with zero velocity at the end of the first burning phase. It then falls to an altitude at which the engines are restarted, resulting in zero velocity at the ground.

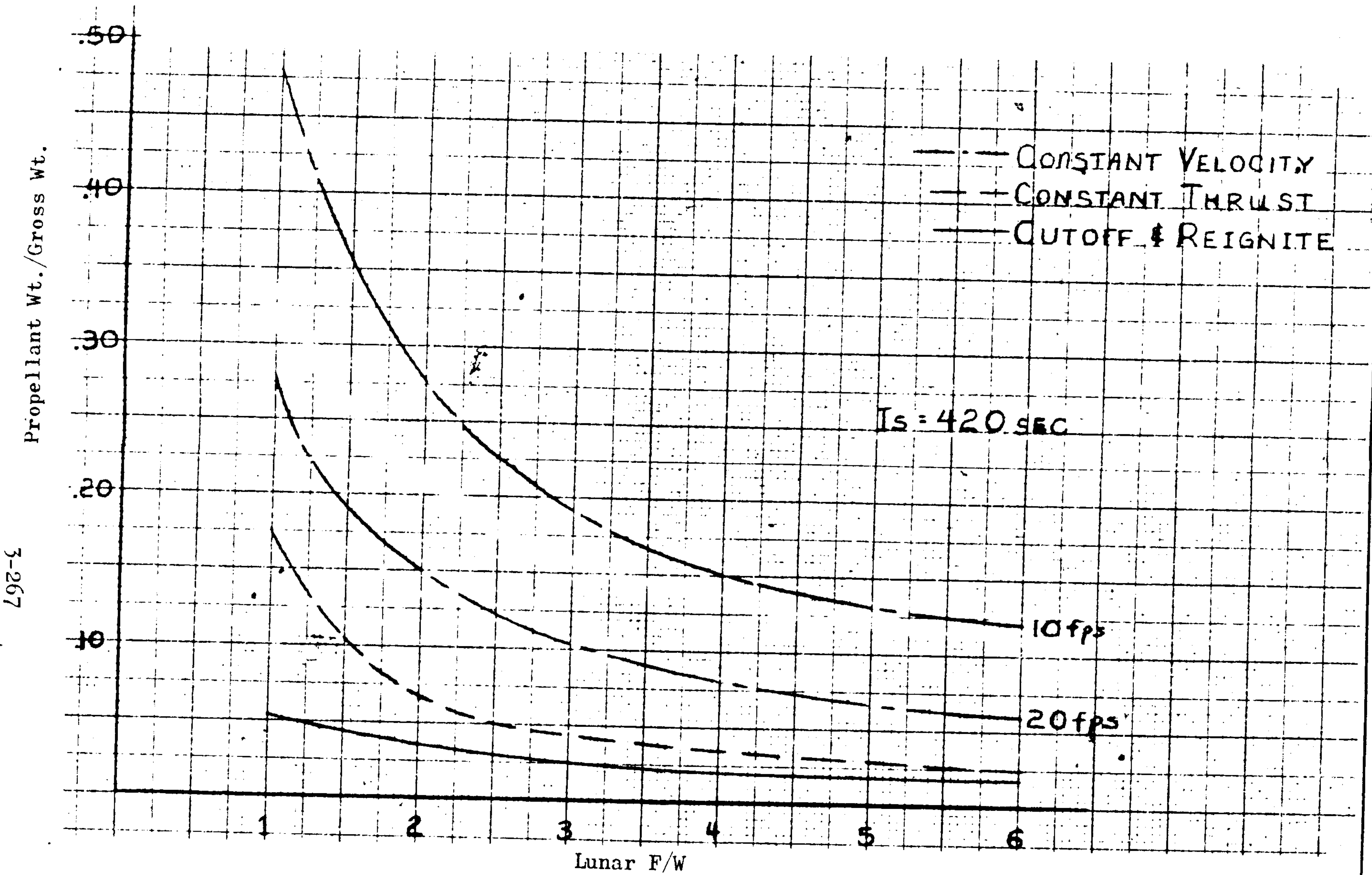

Figure 3-106. Propellant Consumption During Second Landing Phase vs Initial Phase f/w

This altitude is a function of the thrust/weight ratio and altitude at first-phase burnout. Figure 3-107 shows the time required to fall to the second firing altitude as a function of thrust-to-(lunar) weight ratio at the beginning of the first firing phase. Figure 3-108 shows the altitude at which the second firing is initiated.

Again, the effect of measurement errors must be considered. Figure 3-109 gives the altitude at which zero velocity occurs if the second firing altitude and velocity were again 0.1 percent high and low respectively. Designing the possibility of this error into the system will result in the maximum second burnout altitude being twice that shown in Fig. 3-109. This altitude is low enough so that the vehicle can free fall and not impact with a large velocity as long as the initial thrust to weight ratio is sufficient. If a limit is imposed on the impact velocity it follows that a lower limit will be placed on the initial thrust-to-weight ratio. Examination of Fig. 3-109 and 2-130 results in the conclusion that if an impact velocity of as much as 20 fps could be tolerated then the initial lunar landing thrust-to-weight ratio is restricted merely to values above 0.75 lunar g (0.12 Earth g). This is, in reality, no restriction since mass ratio optimizations will result in a thrust-to-weight ratio appreciably higher than this value.

Figure 3-110 shows the total mass ratio required for both maneuvers based on a specific impulse of 420 sec, and indicates a slowly increasing mass ratio requirement down to thrust/weight ratios of approximately 2. Below this value the mass ratio begins to rise rapidly because of the prolonged burning times and large error possibilities.

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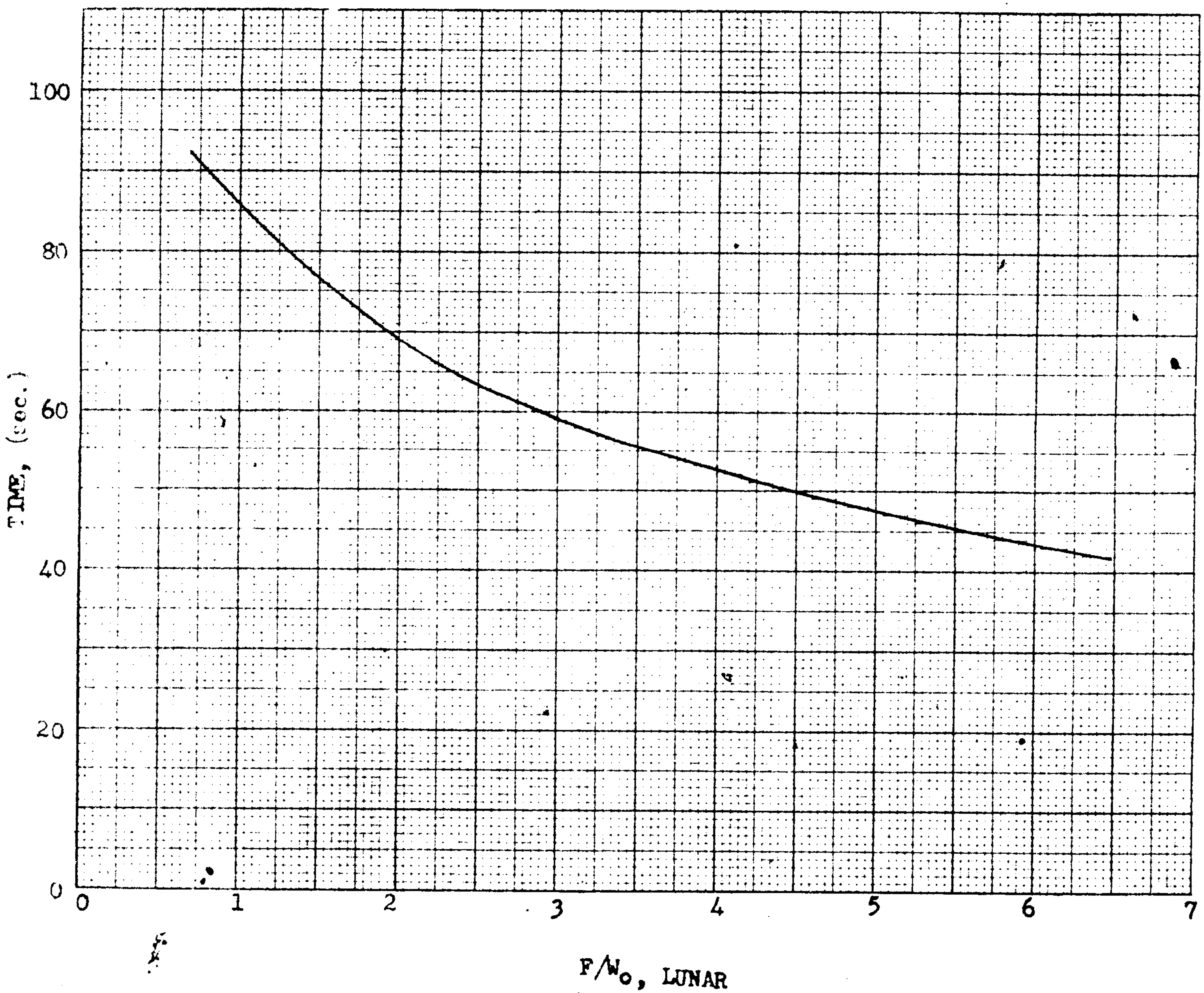

Figure 3-107. Free Fall Time Between Propulsion Phases vs Initial Lunar F W

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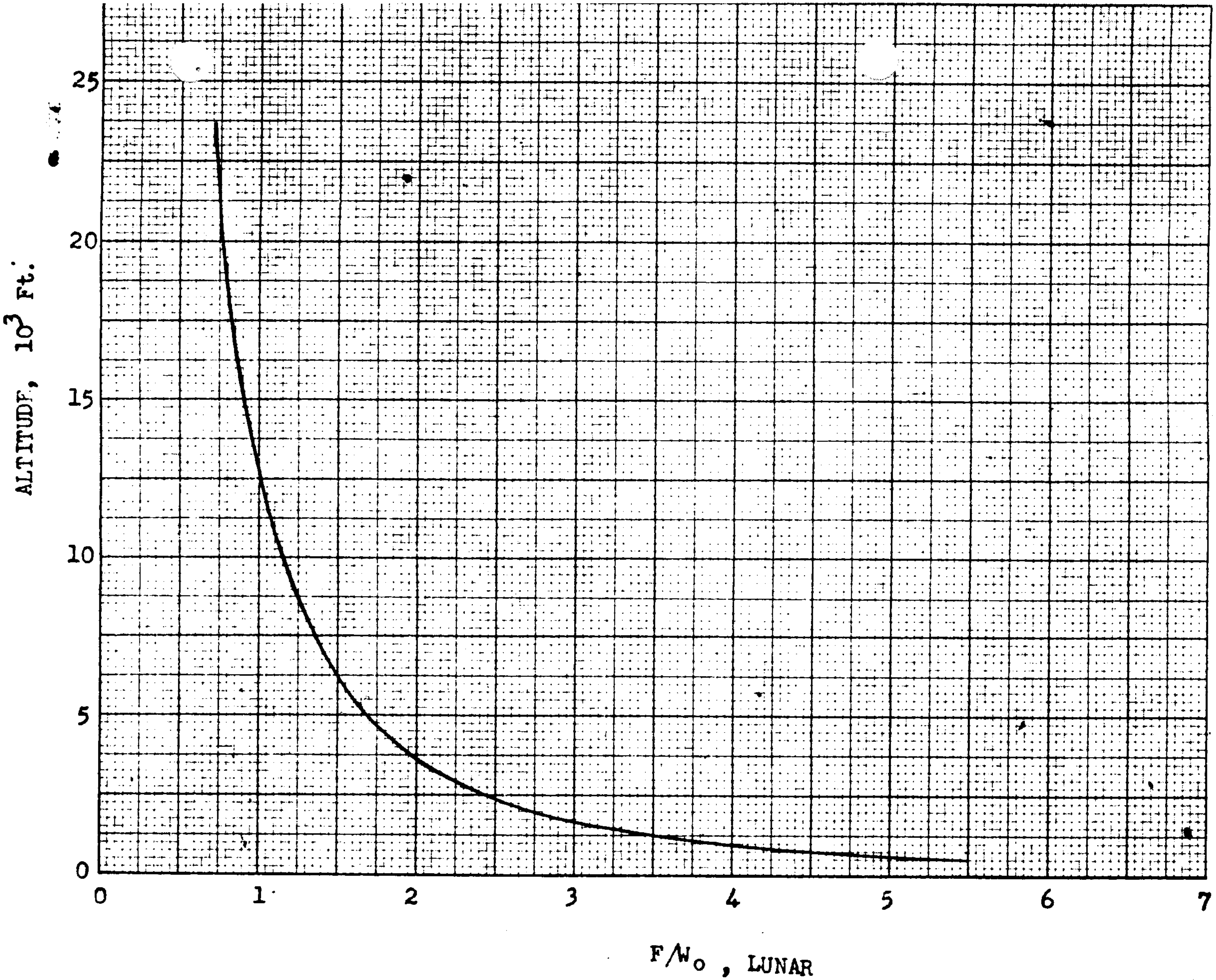

Figure 3-108. Second Firing Altitude vs Initial Phase) F/W

[redacted]

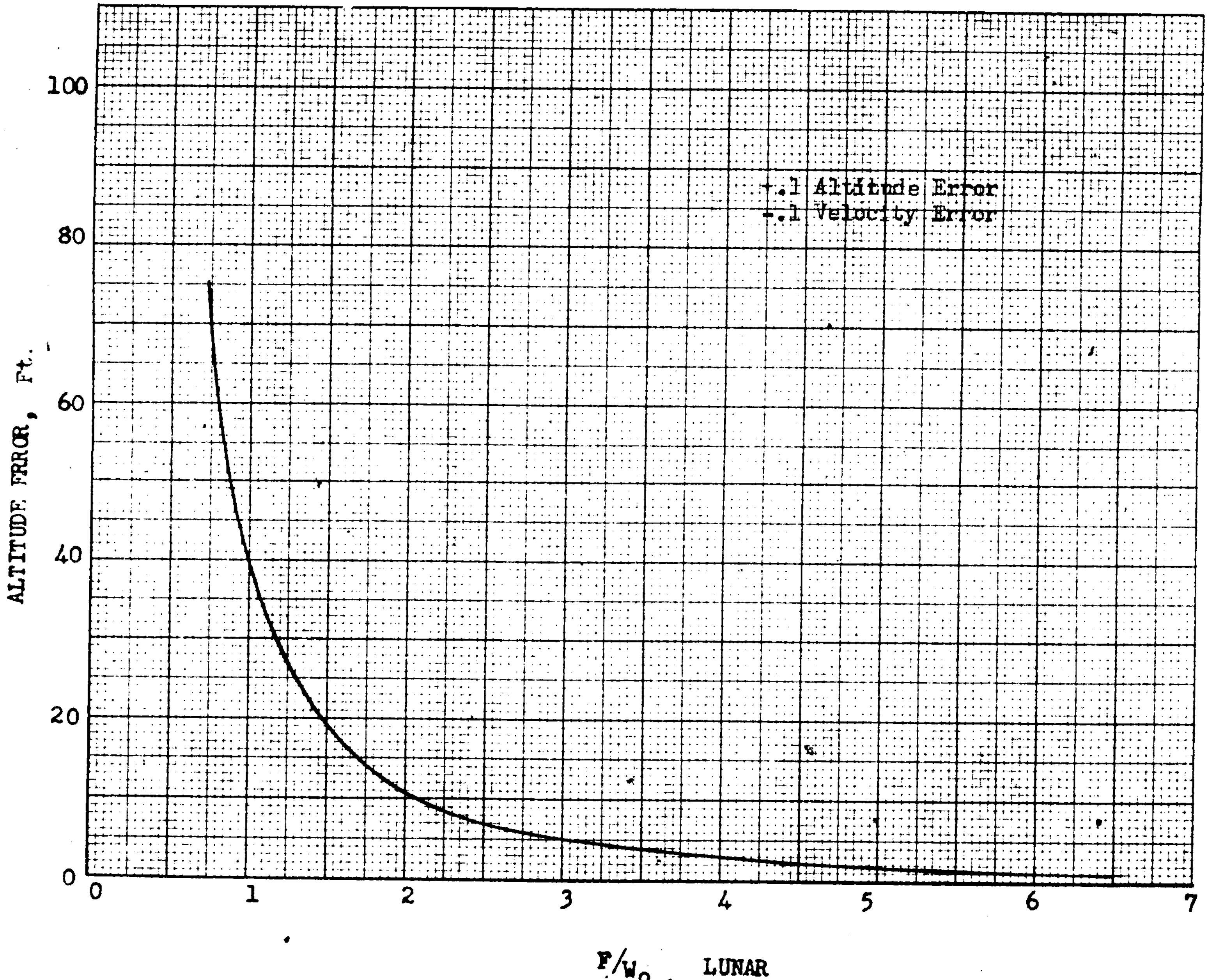

Figure 3-109. Second Altitude Error vs Initial (First Phase) F/W

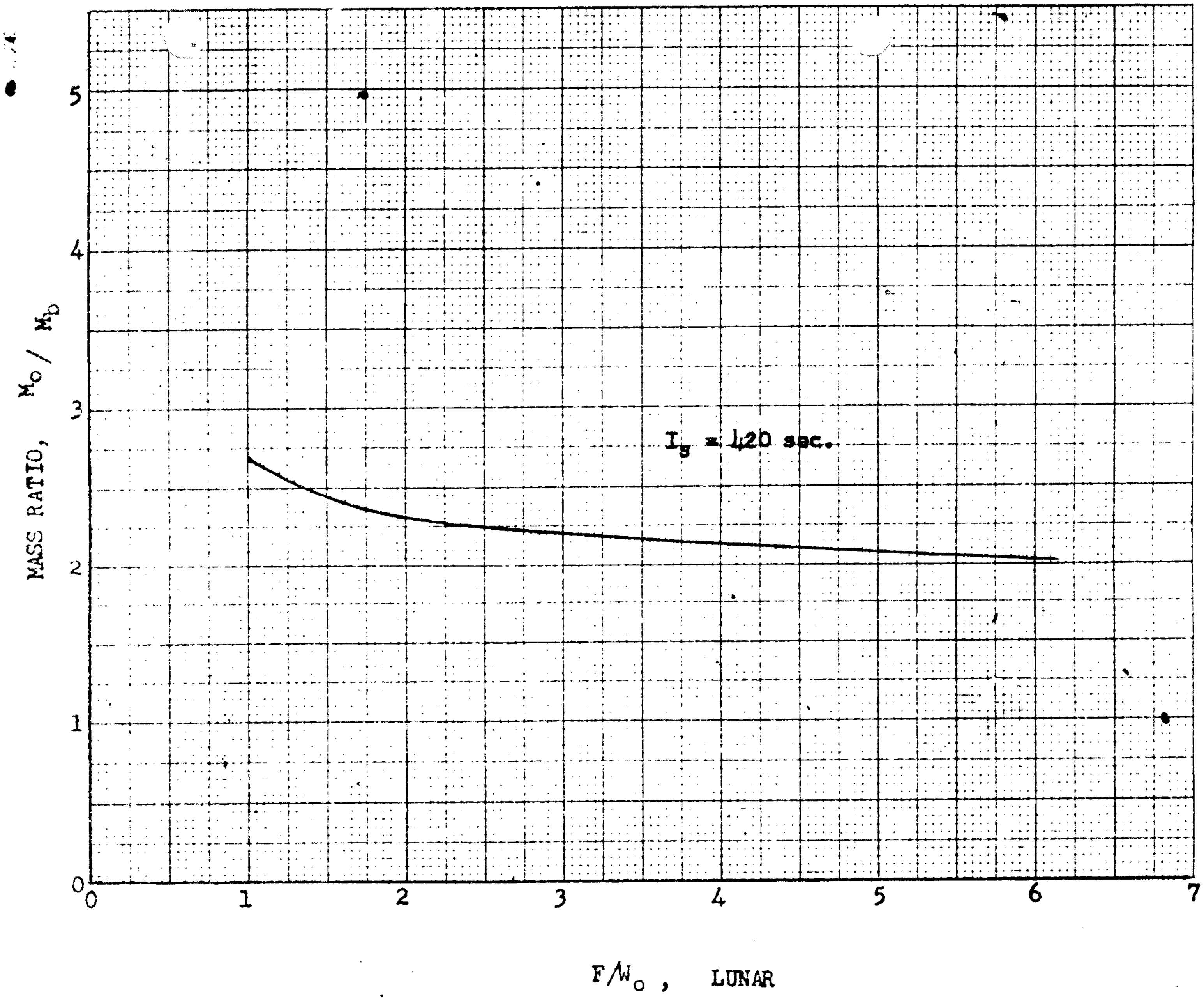

Figure 3-110. Total Mass Ratio vs Initial (First Phase) F/W

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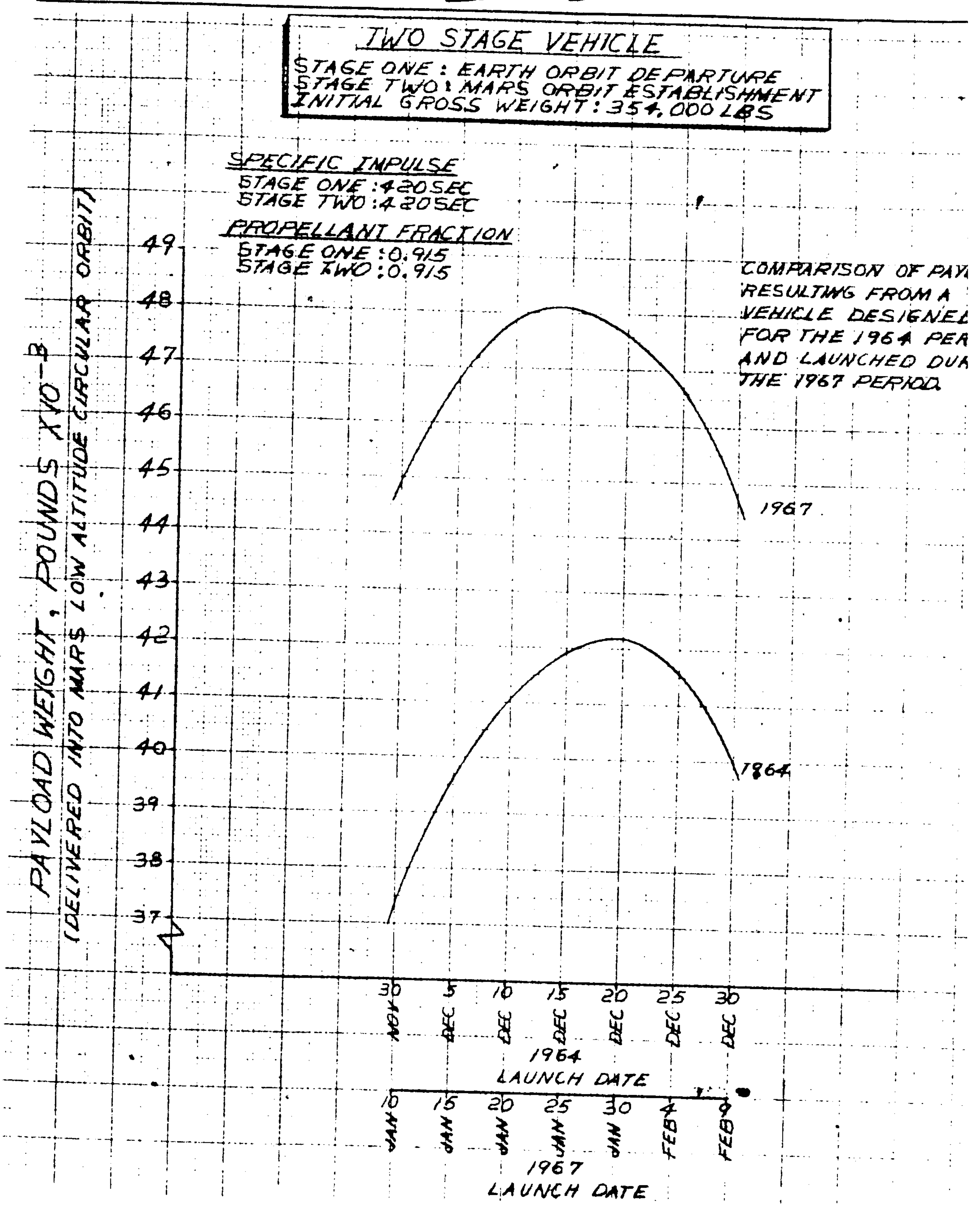

Figure 3-96. 200 Day Mars Transfer, Comparison of Payloads

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EARTH ORBIT RENDEZVOUS MISSION

By contractual agreement, the major efforts of this study were devoted to the Mars and lunar missions. Consequently, the rendezvous analysis was directed toward a single mission: one contact in a 300 n mi orbit, with provision for plane change. While this mission is typical of rendezvous requirements, the conclusions could be affected by significant variations in such parameters as orbit height, gross weight, number and position of contacts, and relanding requirements.

TRAJECTORY ANALYSIS

Many techniques are available for the accomplishment of the rendezvous mission as suggested in Fig. 3-97 . Section 2-3 of the report indicates that the propellant requirement for establishing rendezvous with a non-evasive target in orbit is only slightly greater than the requirement for placing the vehicle in the same orbit without rendezvous considerations. In this study, it is assumed that the rendezvous is to be used for such purposes as orbital buildup, space station supply, etc. The maneuver selection is to be based, therefore, not primarily on propellant consumption optimizations, but rather on the demands which the maneuvers place upon the engine and guidance systems. Emphasis will be given to the effects on engine parameters, but results of the literature search of Phase 1 concerning guidance requirements will be included and considered where applicable.

The basic trajectory is presented in Fig. 3-98 , and the maneuvers are listed in Table 3-11 . The ascent suggested is the conventional vertical rise,

MANEUVER COMBINATIONS FOR EARTH SATELLITE RENDEZVOUS

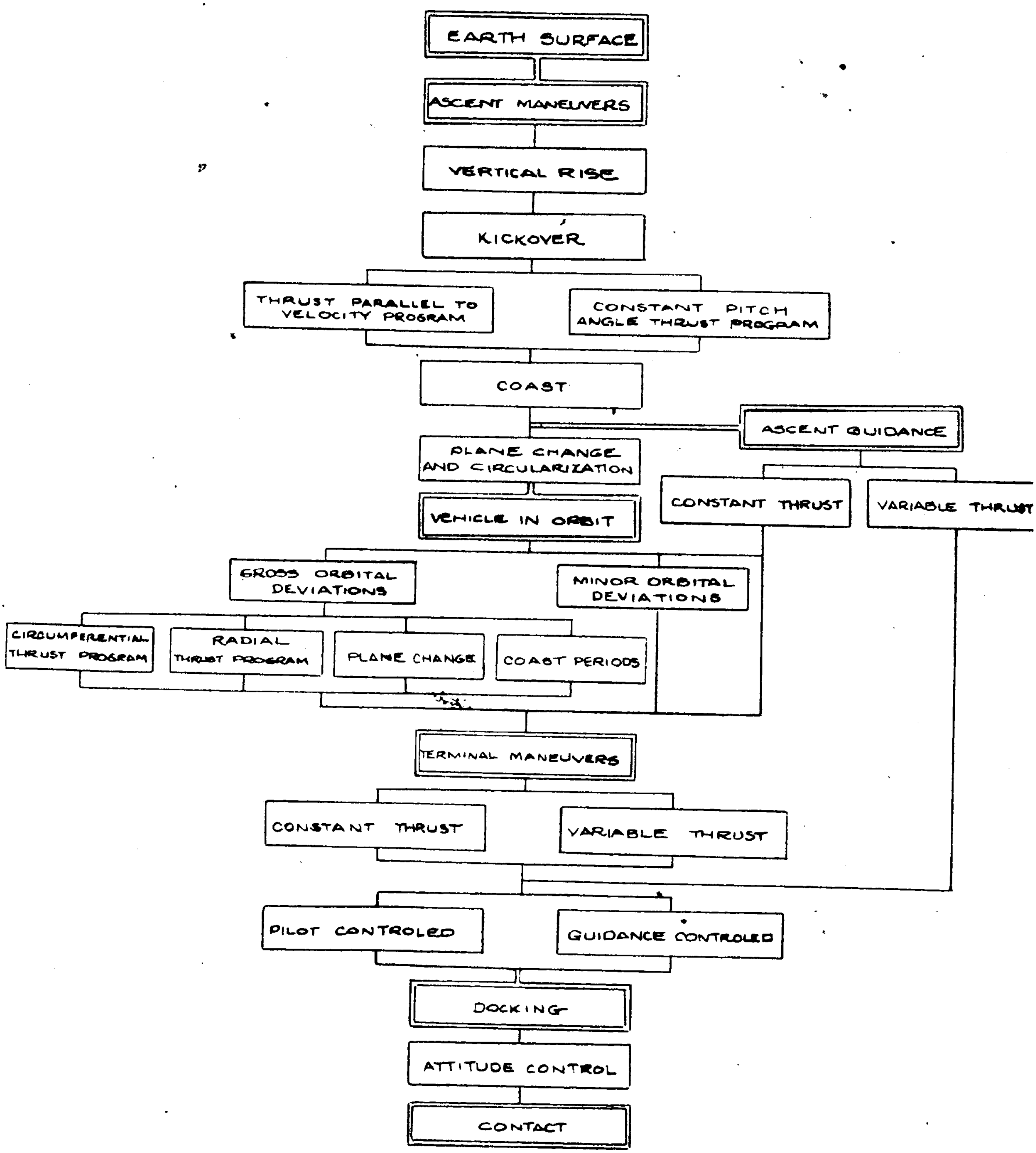

Figure 3-97. Maneuver Combinations for Earth Satellite Rendezvous

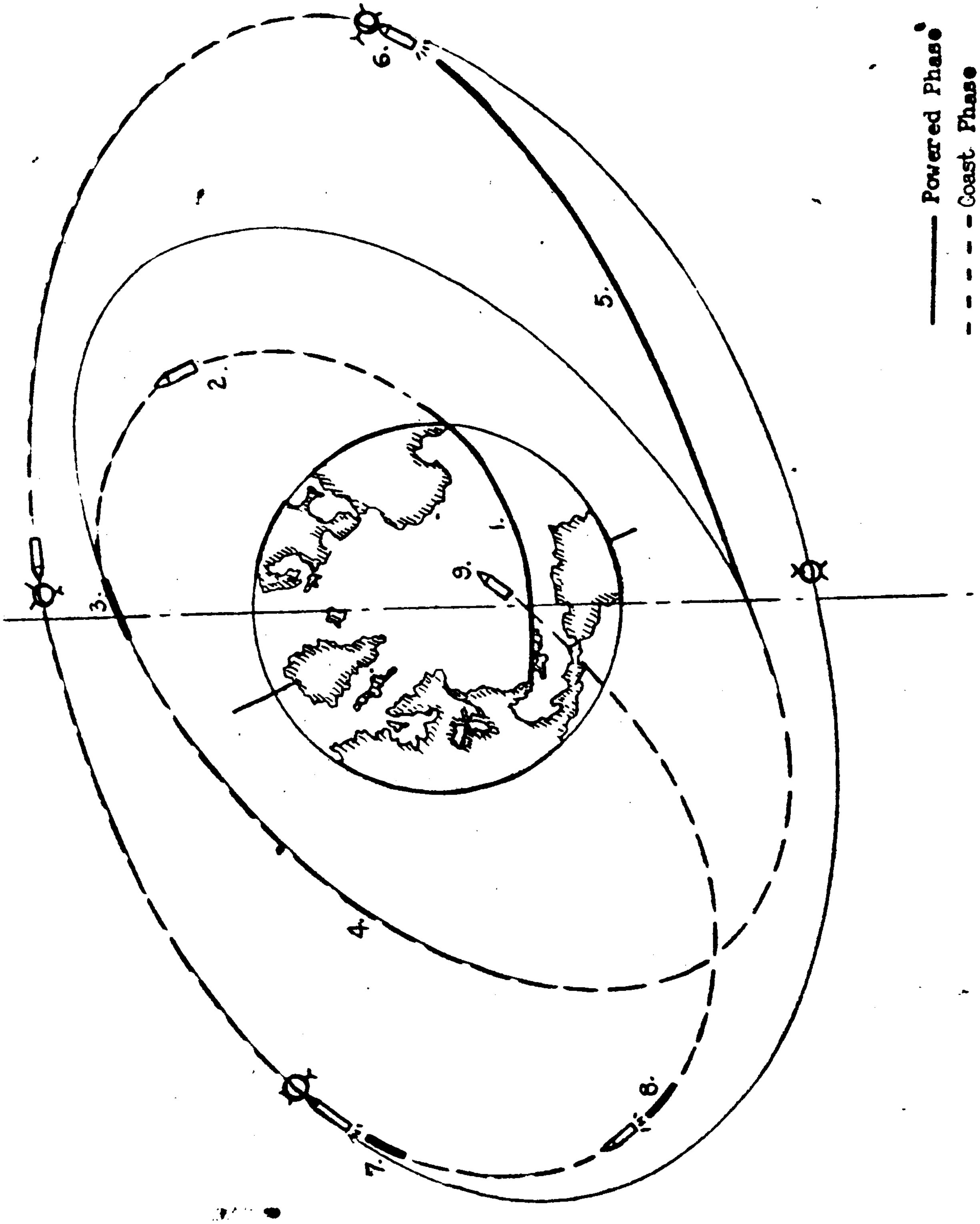

Figure 3-98. Basic Trajectory

TABLE 3-11

MISSION TRAJECTORY MANEUVER PHASES

1. Powered Booster Phase
2. Coast to Orbit
3. Powered Orbit Injection Phase
4. Orbital Coast Phase
5. Powered Plane Change and Rendezvous Intercept Phase
6. Powered Rendezvous Closing Phase
7. Powered Retrothrust Re-entry Phase
8. Powered Re-entry Correction Phase
9. Aerodynamic Re-entry and Landing

kickover (velocity vector rotation) and thrust-parallel-to-velocity maneuver. This continues until dynamic pressure decays below a predetermined value based on drag and missile structural considerations. At this point thrust is applied at a constant angle with the local horizontal until the vehicle position and velocity vectors are such that thrust is terminated and the vehicle coasts to the desired orbit height.

At the apogee the vehicle does not possess sufficient velocity to follow the orbit desired, and further thrust must be applied. In general a plane change is also required. This is most efficiently accomplished by launching so that the vehicle crosses the target orbit plane about a quarter revolution after launch. For circularization of a 300 n mi orbit and a 5-deg plane change the velocity increments are 450 and 2200 fps, respectively. The apogee and intersection points are generally not coincident.

Three alternatives present themselves at this point. The booster engine may be reignited at apogee to provide the circularization velocity increment, and again to provide the plane change increment, leaving only a small residual closing velocity (approximately 100 fps) to be counteracted by the space engine. Secondly, the booster may provide the circularization velocity while the space engine accomplishes the plane change and rendezvous. As a final alternative, the space engine may be used to provide propulsion for circularization, plane change, and rendezvous. The factors to be considered in deciding which alternative to pursue are payload, reliability, and guidance requirements.

The payload capabilities of the latter two methods are compared in an example given as the appendix of this section where it is concluded that allowing the space engine to perform more of the mission results in a slight though not too significant payload increase due to more optimum staging conditions. The payload resulting from the first alternative would be smaller than those of the latter two due to the even further off-optimum staging. The third alternative appears slightly more desirable from a payload standpoint.

The number of starts of each propulsion system are tabulated below for each of the alternative staging configurations.

Staging	Starts: Booster	Starts: Space Engine
I	3	1*
II	2	1*
III	1	2*

Thus, the first alternative requires four starts, while the second and third require only three starts. Also, since the start-and-run reliability of the smaller, storable propellant, pressure-fed space engine would generally be higher than that of the large cryogenic, pump-fed booster (Ref.14), the reliability of the system employing the third alternative appears to be highest.

Turning to the question of guidance system requirements, the 300 n mi orbit with 5-deg plane change may again be used as an example. The initial closing velocity (2200 fps) corresponds approximately to the plane change velocity increment for all alternatives. The high thrust-to-weight ratio (F/W) of the booster propulsion system used in the first alternative permits

*These are the minimum number of starts. The actual number depends on propulsion and guidance systems design.

target aquisition and firing to be delayed until the separation between the target and vehicle is in the order of a few miles. The low F/W (0.05 to 0.2) required of the space engine for the final closing maneuvers dictates a target vehicle range in the order of 100 n mi at target aquisition. Based on Ref. 15 , radar sensors for this range can be designed at an approximate weight of 50 lb. exclusive of power supply.

In conclusion the maneuver employing the space engine for all phases appears most promising from payload and reliability considerations, and lies within the capabilities of guidance sensor devices.

Landing from orbit can be accomplished by providing a 500 fps deorbiting velocity decrement followed by aerodynamic re-entry techniques.

PROPULSION SYSTEM

Figure 3-97 presents several concepts of propulsion and guidance for the terminal phase Throttleable or constant-thrust engines are suggested, and piloted and guidance control are presented for selection.

The findings of the literature search were that the type of control (piloted vs automatic) does not greatly affect the propulsion requirements. Thus the selection of the type of control will be left to other considerations.

The advantages of both the throttleable and the pulsed engines must be weighed carefully before deciding which is most applicable. The throttleable engine has the advantages of requiring only a single start; of providing fine control if deep throttling is used; and of having the capability of applying high thrust under emergency conditions. Despite these advantages the fixed-thrust engine system was selected on the basis of its one primary advantage: simplicity. To avoid extremely deep throttling requirements the variable thrust system would have to employ a two-engine propulsion system (Ref. 15). This would consist of a relatively high-fixed-thrust engine and a lower-thrust level engine capable of about 3 or 5:1 throttling. The complexity of this system would result in lower reliability than that of the multistart fixed-thrust engine.

When the selected system is used to rendezvous payloads of the Nova H-2 class vehicle, a fairly large propulsion system (approximately 12,000-lb-thrust level) results. This prohibits using the rapid pulsing sequence employed by the low-thrust engines. Thus, after accomplishing the plane change and allowing a small residual closing velocity, the system would operate by applying a small number of relatively long-duration impulses (in the order of 1 to 50 sec). The velocity increment imparted by the impulses is shown in Fig. 3-99 . A brief analysis based on the tracking and guidance accuracies of Ref. 15 indicated that four firing periods would be generally adequate to rendezvous at a closing velocity of less than one fps. Velocity errors due to cutoff impulse uncertainties are insignificant when operating at this thrust level.

A three-axis attitude control system using twelve nozzles may be used to control the vehicle orientation during the rendezvous phase and to accomplish the final docking phase after the relative velocity has been reduced to less than one fps by the main rendezvous propulsion system.

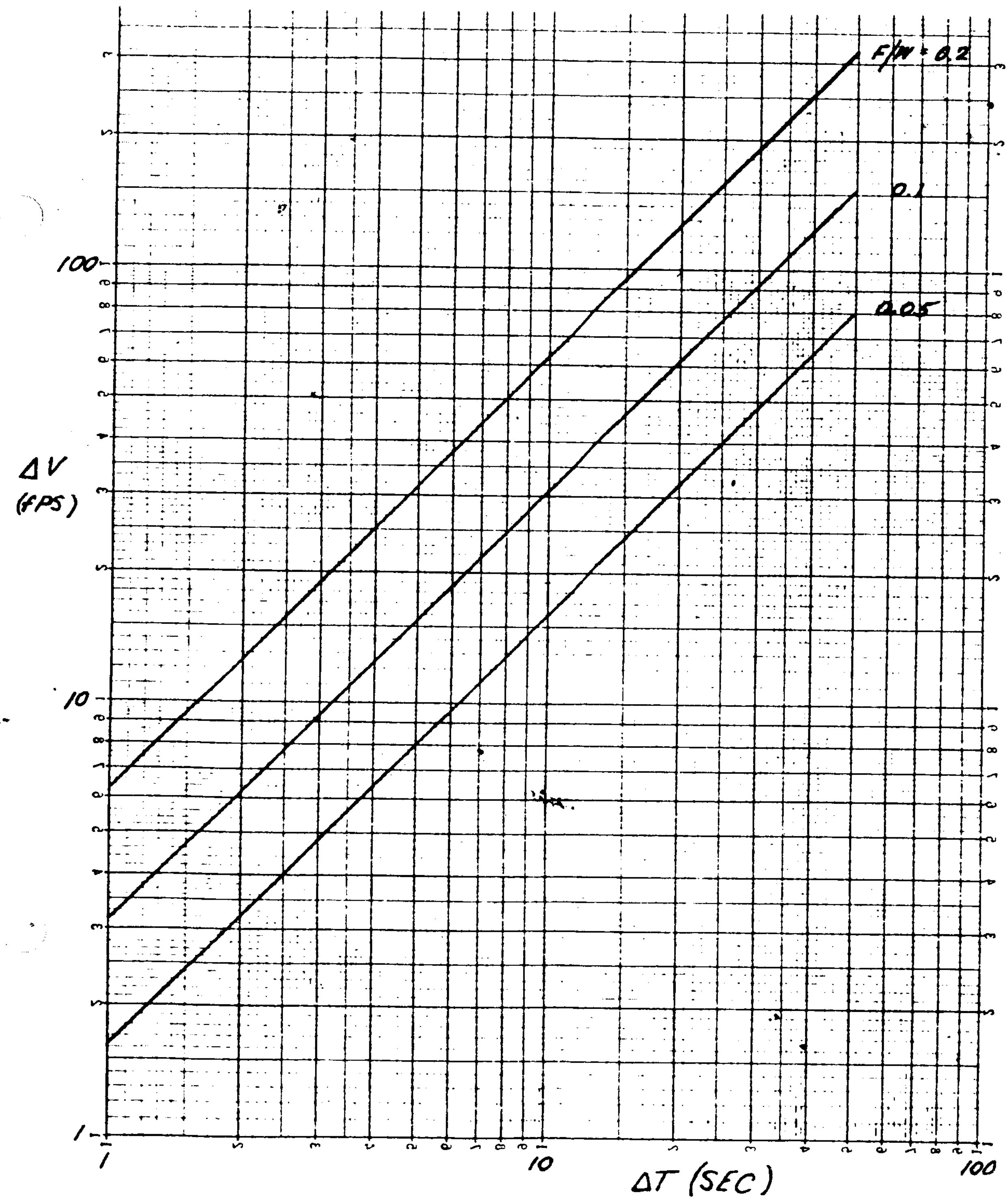

Figure 3-99. Velocity Increment vs Pulse Width

Just as the maneuver selection was based on factors other than payload, due to the small velocity increment involved, so also is the type of propulsion system dictated by such factors as simplicity and reliability. Ref. 14 contains a table of reliabilities of engines of about the same thrust level as that of the rendezvous engine. The variables of size, propellants, feed system, and pressure are included. The propulsion system envincing the highest reliability was the small pressure-fed, storable propellant system and was therefore selected as the rendezvous propulsion system. Selection of hypergolic storable propellants and positive expulsion tanks is also attractive from considerations of restart capabilities. Table 3-12 describes the recommended rendezvous propulsion system. No specific amount of propellant has been allocated for the landing maneuver due to the wide range of possible requirements, i.e., from no landing through landing the entire payload. However, it has been calculated that 150 lb additional propellant would be required to deorbit and land the empty stage with aerodynamic braking. The propellant requirement increases to 4800 lb to land the orbited payload.

OPERATIONAL ASPECTS

Significant to the efficient accomplishment of the mission are the operational methods employed. This is particularly so in the orbital buildup missions. For example, if several payloads are to be joined in orbit, the launch vehicles should be identical when possible so that a separate standby vehicle does not have to be available for each launch vehicle. The same philosophy may also be extended to the payloads (e.g., identical propellant carrying tank configurations for orbital fueling).

TABLE 3-12

RENDEZVOUS PROPULSION SYSTEM

Payload, lb	92,800
Propulsion System	
Feed System	Positive Expulsion
Propellants	MON/MMH
Propellant Weight, lb	28,400
Inert Weight, lb	3200
Thrust, lb	12,000
Restarts	~ 3

[redacted]

The significant quantity of propellant indicated in Table 3-12 is due, for the most part, to the plane change phase of the rendezvous. The 5 deg plane change capability is included, primarily to allow rendezvous of any of five passes of a target satellite launched from the same site. Operational considerations may allow reduction of this angle to 2 deg or less and thus significantly affect the propulsion requirements.

Booster recovery has received attention in the past, but should be further emphasized for operations requiring many launches to complete the mission. The long engine life of a liquid rocket propulsion system and the small percent of the total cost involved in refueling the vehicle make recovery quite attractive.

For rendezvous involving several payloads to be assembled by Man in orbit, operational analyses would have to be carried out to determine the optimum time (i.e., which launch) to place the man in orbit. Other operational problems occur which may be general, or applicable only to specific situations, but are of prime importance in the planning of any rendezvous mission.

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APPENDIX A

STAGING EFFECTS ON ORBITAL PAYLOAD

The trajectory used for establishing a low Earth orbit (a parking orbit or a space mission) generally includes a coast phase followed by a circularizing maneuver propulsion phase to add the final velocity needed to establish the orbit. This final orbit establishment propulsion phase can be supplied by reignition of the last stage of the boost vehicle, or it can be supplied by the first space stage, which would later be reignited for the next propulsion phase. The performance of an example vehicle was evaluated to illustrate the effects of these two alternate approaches. The vehicle selected consisted of an O_2/RP propellant booster, an O_2/H_2 propellant second stage and a space stage using MON/MMH propellants. The mission selected for the vehicle was the placing of a payload into a 300 n mi orbit with the inclusion of a 5-deg plane change. The boost vehicle will deliver 118,000 lb (space stage plus payload) to a 300 n mi Earth orbit, and will deliver 124,400 lb (space stage plus payload) to the coast phase of the parking orbit trajectory. After the coast phase of a parking orbit trajectory, about 454 ft/sec must be added to the vehicle velocity to establish the orbit. Figure 3-100 shows the ideal velocity requirement needed to provide the required 454 ft/sec of additional velocity as a function of the vehicle thrust-to-weight ratio at the initiation of the orbit establishment phase. The data are presented for a system having an I_s of 420 sec, but would not be significantly different for a storable propellant combination due to the small velocity increment. If the second stage of the boost vehicle were used for this orbit establishment phase, the thrust-to-weight will be high; for this vehicle about 4.7.

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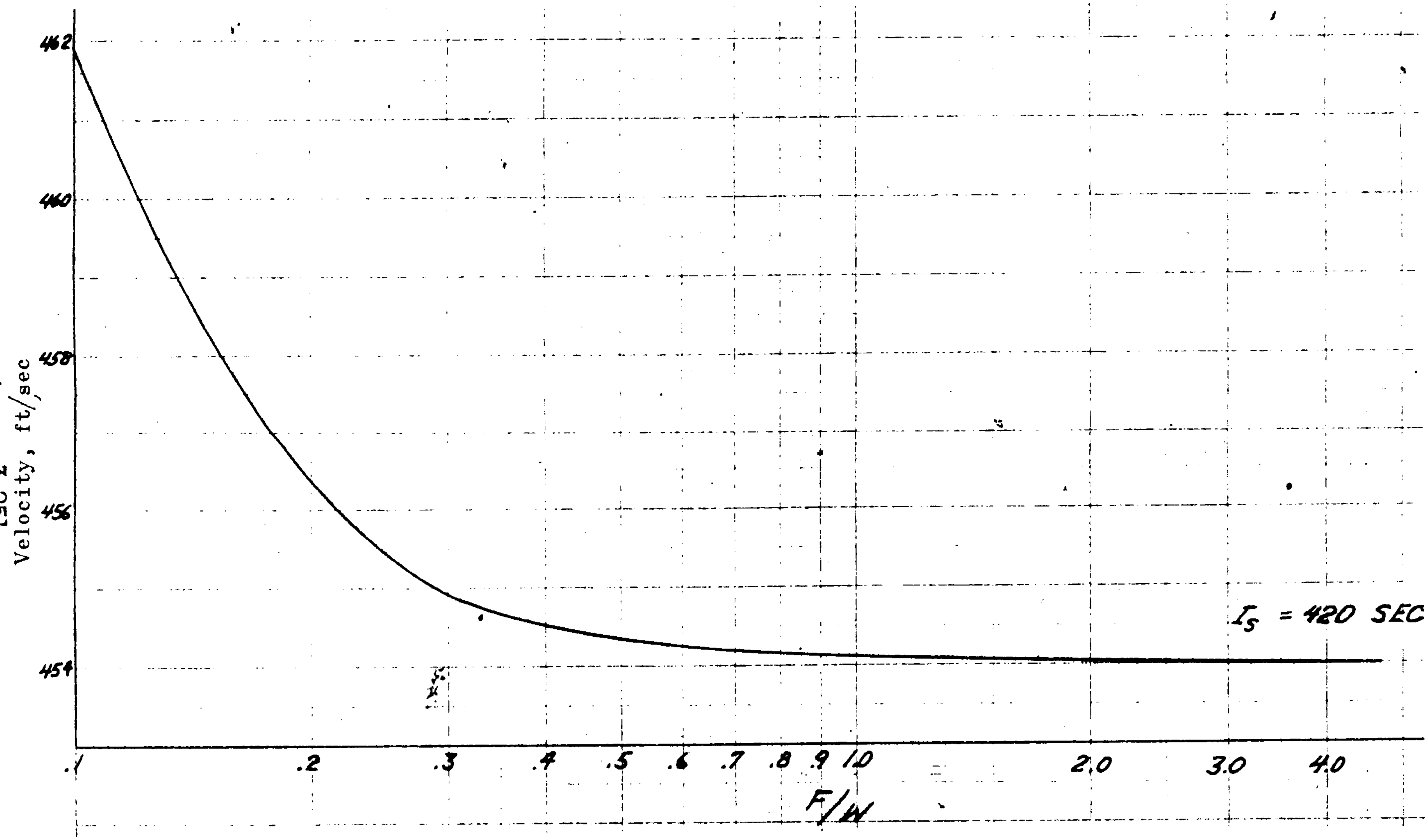


Figure 3-100. Ideal Velocity Required for 300 n mi Orbit Circularization vs Initial Thrust/Weight Ratio

[redacted]

The space stage used for the Earth departure phase will have a considerably lower thrust-to-weight (about 0.1); however, Fig. 3-100 shows that this lower thrust-to-weight will affect the ideal velocity requirement for orbit establishment only slightly. A thrust-to-weight of 0.1 requires only about 8 ft/sec higher ideal velocity than a 4.7 thrust-to-weight. Even though the ideal velocity requirement for orbit establishment is essentially the same whether the 4.7 thrust-to-weight second stage of the boost vehicle or the 0.1 thrust-to-weight space stage is used; the weight in orbit, and consequently the thrust-to-weight at the initiation of the plane change phase, will be different for a space stage with the same thrust (approximately 12,000 lb thrust) in both cases. The difference in weight prior to the initiation of the plane change maneuver, however, is small (118,000 lb when using the second stage of the boost vehicle for orbit establishment and 120,200 lb when using the space stage for orbit establishment). Figure 2-8 shows that this small change in thrust-to-weight will cause less than 10 ft/sec difference ideal velocity required to accomplish the 5-deg plane change. In evaluating the payload delivered, the inert weight of the space stage was calculated as follows:

Guidance and Control Weight, lb = 500

Engine Weight, lb = 230

Tank and Structure Weight, lb $= 0.813\ W_p + 0.219\ (W_p)^{2/3}$

The payload delivered by the vehicle using the second stage of the boost vehicle for orbit establishment was 92,500 lb, and that delivered by the vehicle using the space stage for orbit establishment was 92,800 lb. From the above illustration it seems likely that payload will not be substantially affected by using the space stage rather than the last stage of the boost vehicle for the parking orbit establishment for most vehicles and space missions. The advantages in payload derived from

Because the vehicle is decelerating continuously as it descends the question arises as to whether appreciable translation could be accomplished to avoid ground obstacles if 10,000 ft is the altitude limit beyond which useful resolution of the desired quality cannot be accomplished. The time spent below this altitude for the single firing phase is presented in Fig. 3-111. Within the range of expected thrust/weight values (2 to 5) it appears that only 20 to 30 sec are available to obtain pictures of the surface, transmit to Earth, and maneuver the vehicle. Because the time required to transmit the picture is approximately 20 sec, based on Ref. 24, this correct-while-descending technique is clearly not feasible for this trajectory.

Two alternatives are presented: the first is to design a hovering capability into the vehicle; and the second is to select a landing spot, based on previous orbital reconnaissance, in which the probability of an obstacle encounter would be low. Because a guiding philosophy of this trajectory was that the guidance and propulsion systems be as simple as possible, it was decided to eliminate the engine throttling or excessive (Fig. 2-29) gimbaling requirements, as well as the accompanying optical and guidance systems, in favor of the latter alternative.

Intermediate Orbits

Later lunar missions may be characterized by having manned payloads, improved guidance systems, more sophisticated propulsion, and an objective of landing at an exact predetermined spot (possibly for rendezvous) on the surface of the Moon. The nature of the payload (manned) dictates that the probability of returning to Earth safely

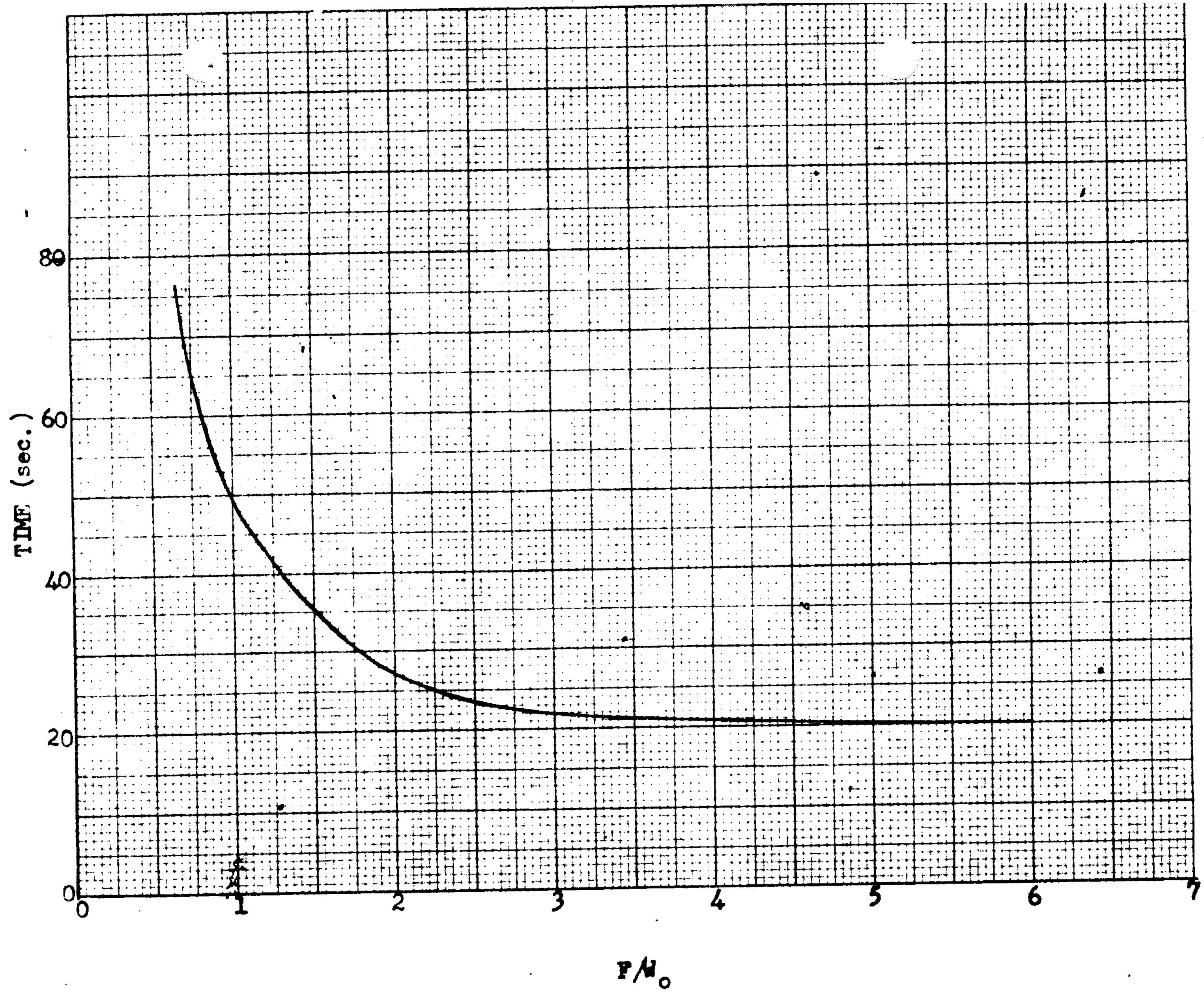


Figure 3-111. Duration Below 10,000 ft vs Initial (First Phase) F/W

[redacted]

be extremely high even if the mission itself were not accomplished. Thus, one of the guides for the trajectory selection for this mission was that, whenever possible, system failure would result in the spacecraft traveling in an orbit around the Earth or the Moon, with the possibility of repairing the malfunction, returning to earth, or at least waiting for a rescue vehicle. The maneuvers selected to accomplish this objective are presented in Fig. 3-112.

25

Reference 25 indicates that launch abort systems for manned spacecraft will be needed to cope with abort problems during the following phases of the launch sequence: (1) on the launch pad and during lift-off, (2) during maximum dynamic pressure, (3) during suborbital flight, and (4) during superorbital flight. The propulsion requirements for (1) and (2) are somewhat similar and have the objective of propagating as much distance as possible, in a short time, between the payload capsule and the booster vehicle. The requirements of the second phase are greater than for the initial phase because the capsule is more strongly affected by drag than the remainder of the vehicle because of its low length-to-diameter ratio and consequent small ballistic parameter. This means that the capsule would tend to decelerate more rapidly than the booster so that a relatively high thrust/weight ratio must be used. Because these rockets would be attached to the capsule and jettisoned after leaving the atmospheric drag region, it follows that the thrust-to-weight ratio should be based on the weight of the capsule. Assuming a dynamic pressure (P) of approximately 600 lb/sq ft based on anticipated trajectories, and an approximate capsule weight (W) and diameter (D)

[redacted]

MANEUVER COMBINATIONS FOR LUNAR LANDING

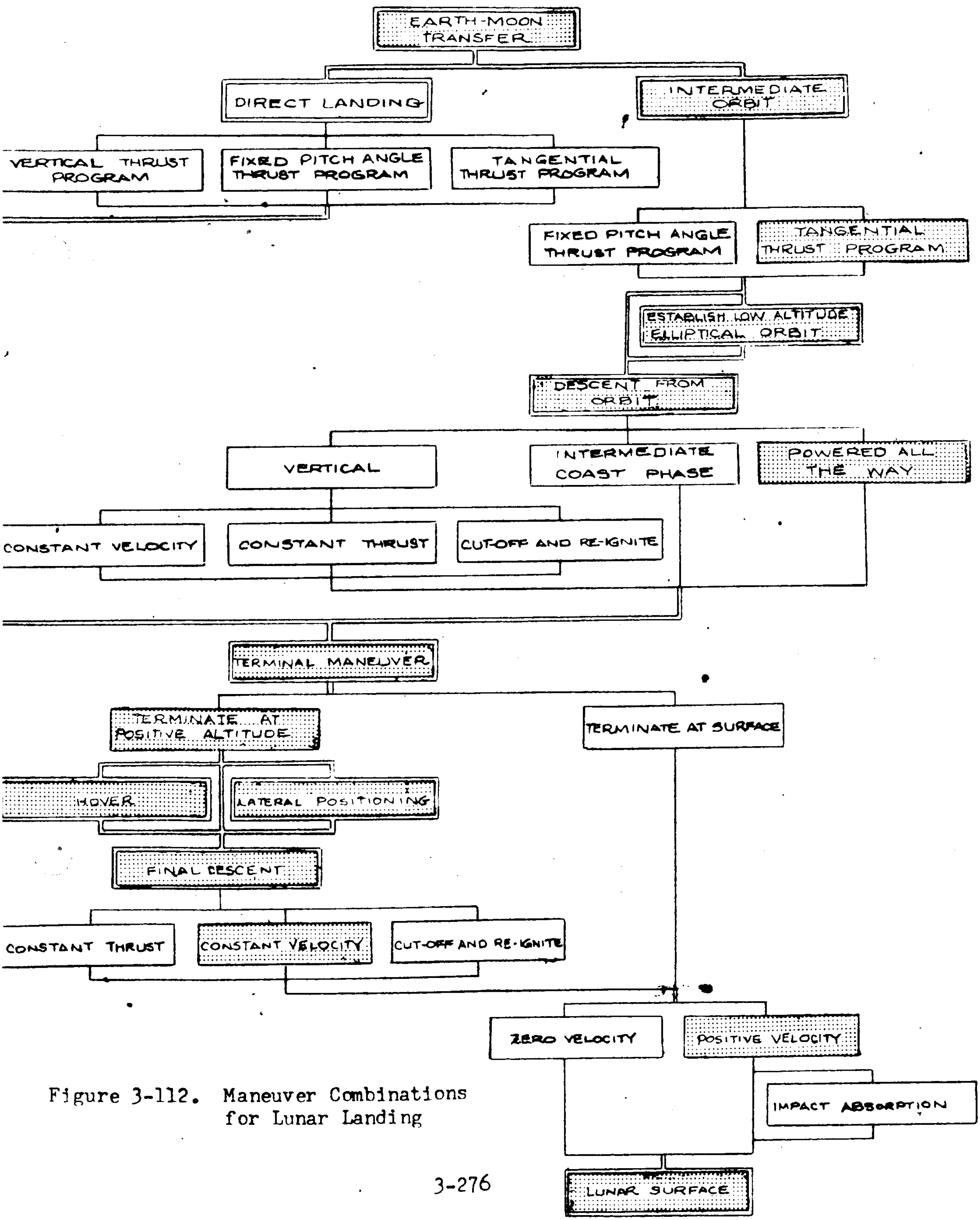


Figure 3-112. Maneuver Combinations for Lunar Landing

[redacted]

of 10,000 lb and 10 ft respectively, based on Ref 26, the deceleration due to drag is

$$g_d = \pi D^2 P/4W = 4g$$

The drag deceleration of the booster will be inversely proportional to the ratio of booster to capsule weights. Therefore the deceleration of a 1,000,000 lb booster is only 0.045 g; a negligible quantity by comparison. If it is desired to obtain a separation (S) of 5000 ft between the capsule and the booster within 5 sec after abort, the net acceleration (considered constant) must be

$$g_n = 2S/g_o t^2 = 12\ g$$

$$g_o = 32.2\ ft/sec^2$$

The total thrust/weight ratio is then approximately 16.

If this thrust were applied for a launch abort the entire 16 g would be felt by the crew. The most likely method of reducing the acceleration would be to cluster several abort rockets so that only those necessary to give the required acceleration would be fired at any abort time.

Abort during the suborbital phase poses problems of thermal and deceleration loadings during re-entry if the proper trajectory is not flown. The purpose of the propulsion system during this abort phase is to orient the velocity vector at re-entry to minimize the peak deceleration loads.

[redacted]

Because the mission was intended to terminate by aerodynamic re-entry it is assumed that the capsule design is adequate to provide for the heat loads encountered during a suborbital abort. Propulsion velocity requirements are approximately 3000 to 5000 fps for this type of abort and, as there is no necessity for high thrust/weight ratios, the propulsion system of the spacecraft would probably be employed.

Superorbital aborts could result in lengthy traverses through the Van Allen radiation belts and thus present a considerable hazard to the crew. Here again, the propulsion requirement is to maneuver the capsule into the proper entry corridor. A propulsion velocity requirement of approximately 4000 fps, and the ability to use relatively low thrust/weight ratios, again permits utilization of the spacecraft propulsion system for this abort maneuver.

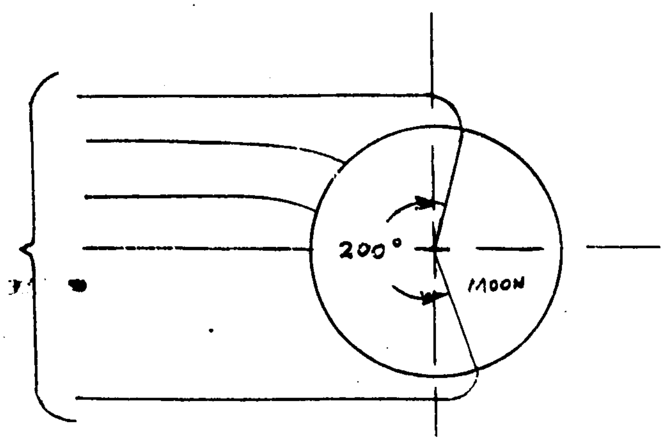
Mission abort may be accomplished any time during the transfer phase by using the propulsion velocity available to place the vehicle in a return trajectory which passes through the proper re-entry corridor. An abort at this time could mean that an emergency has arisen which necessitates return to Earth as soon as possible. In this situation, to obtain the most rapid transfer trajectory, all propellants would be expended, except those required to decelerate the vehicle to a velocity suitable for re-entry. This would necessitate that there be information available concerning the magnitude and direction of the velocity vector to be applied at each point along the outbound trajectory.

The 2.6 day transfer trajectory would again be used in leaving the 300 n mi Earth orbit for the same reasons given in the previous section. Midcourse correction is also assumed to require 150 fps for each direction (outbound and return) of the trip.

In accordance with the philosophy that a noncatastrophic failure should not result in the vehicle crashing, the next trajectory maneuver has been designed to place the vehicle in orbit around the Moon. Thus, if the vehicle passes in front of the Moon and no propulsion phase occurs the vehicle velocity will be retarded to the point where it begins the return trip to Earth along an elliptical trajectory.

There are other advantages of the lunar orbit besides the abort considerations. Prelanding surveys and reconnaissance can be made from orbit if this has not already been accomplished by previous missions. Another significant advantage is the wide range of landing sites possible when using the orbital maneuver. For a given thrust/weight ratio (of at least near-optimum magnitude) a fixed trip time and direct descent using thrust antiparallel to velocity (of which the vertical descent is a special case) results in a choice of landing points along an arc of about 200 deg on the lunar surface as shown in the following diagram.

Constant
time
Earth-Moon
Trajectories



The actual landing point depends upon the lunar position at the time of initiation of the Earth-Moon transfer maneuver. Further discussion of this point is made in Ref. 27.

By use of a lunar parking orbit, landings can be made at any point on the surface of the Moon. The vehicle may be deorbited at any point , the orbit, and the final degree of freedom is obtained by varying the orbital inclination. The inclination may be varied by affecting a plane change during or after the orbit establishment maneuver.

Because this technique is costly from a propellant standpoint, especially if appreciable angles are involved (see Fig. 3-112A), another method is available which allows establishment of orbits which pass over any point on the lunar surface. This maneuver, illustrated in Fig. 3-113 shows several possible orbits obtained by varying by a very small amount the direction of the velocity vector at the transfer injection point. These different transfer trajectories are tangent at different points (A, B, C, D) to the 50 n mi altitude sphere around the moon and thus result in orbits of various inclinations. Note that the orbits are restricted to the extent that they all pass through a common intersection line. Thus, although it is not possible to establish any orbit desired by this method, the allowable orbits do permit landing anywhere on the lunar surface.

The orbit height selected for this mission was 50 n mi. This height is greater than need be from guidance accuracy considerations (which indicate errors as low as 1 n mi after midcourse corrections) but does provide for unexpected gross errors and also is closer to the minimum energy orbit of Fig. 2-104. The propulsion velocity

[redacted]

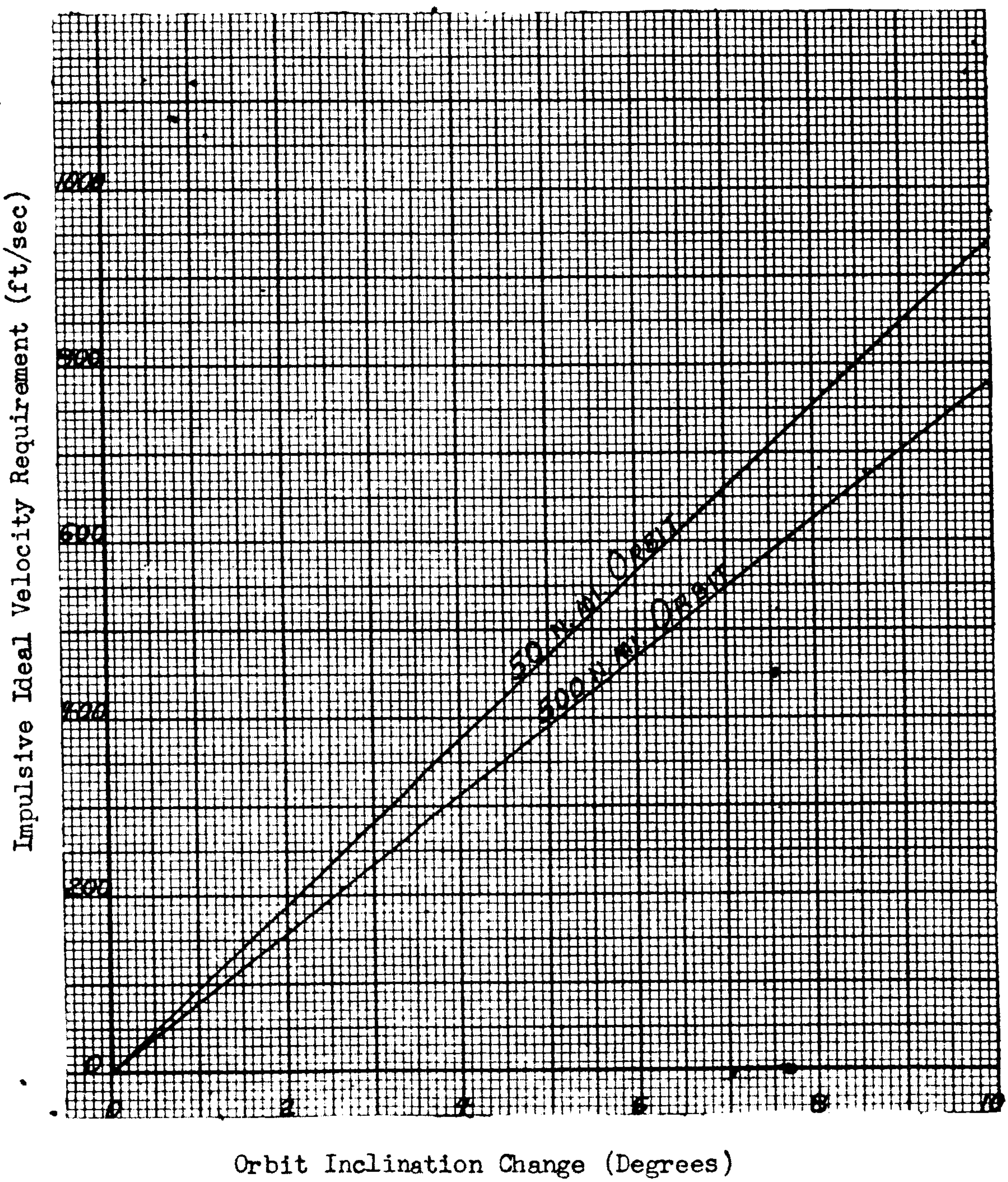


Figure 3-112A, Ideal Velocity Requirement for Orbit Inclination Change

[redacted]

[redacted]

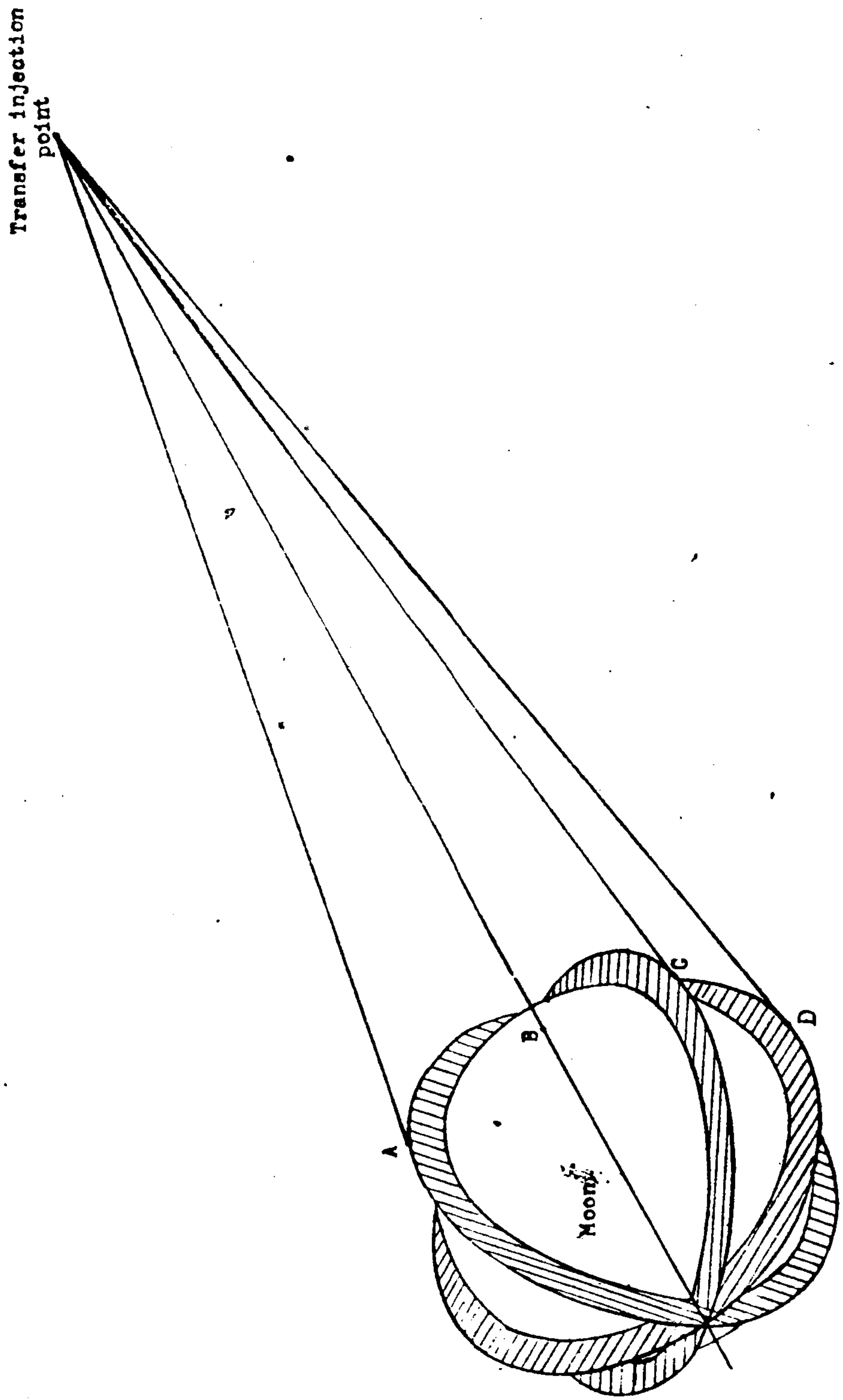


Figure 3-113. Possible Lunar Orbit Inclinations

[redacted]

required to establish the orbit using a thrust parallel to velocity maneuver is shown in Fig. 2-113 and 2-114. The thrust parallel to velocity maneuver was selected on the basis of propellant economy.

An error analysis was made to indicate the amount of added propulsion velocity required to allow for probable deviations in the guidance and propulsion systems. A midcourse guidance system capable of reducing position and velocity errors to 1 n mi and 3 fps respectively at the injection point resulted in an elliptical path deviating only slightly from the intended 50 n mi circular orbit. If the thrust deviated from the design value by -1 percent while the propellant flow-rate remained at the nominal value (i.e., a 1 percent decrease in specific impulse) the resultant orbit would be an ellipse of approximately 75 n mi apoapsis and 50 n mi periapsis. This can be compensated for by expending an additional 30 fps velocity increment at the 50 n mi periapsis. Calculation of the effects of cutoff impulse deviations indicate that the anticipated deviations would result in approximately 0.5 fps velocity errors.

After completing the 50 n mi orbit establishment maneuver and selecting the landing site the vehicle trajectory is converted to a 50 n mi by 30,000 ft ellipse. The perigee of the ellipse is the point from which the final descent maneuver will be initiated. To obtain this ellipse a retrothrust is applied 180 deg around the initial circle from the point at which the perigee is desired. The magnitude of this velocity increment is 60 fps.

[redacted]

The purpose of this maneuver is twofold. Primarily, it is performed to allow the final descent to be made under continuous power without excessive propellant consumption. If a constant thrust descent were made from 50 n mi the velocity requirement would be significantly higher than for descent from 30,000 ft due to the lower thrust level and consequent longer burning time required by the former case.

A secondary purpose of this maneuver is to provide closer observation of the landing area. This is really more of an incidental advantage than an intended purpose since it is probable that adequate reconnaissance would have already been accomplished from the circular orbit.

The height of 30,000 ft was selected as minimum altitude based on knowledge of lunar topography. Higher prominances on the dark side may dictate a higher altitude. Conversely, if it were known that no high areas existed along the orbital path near periapsis, a lower altitude might be selected to reduce the final landing propellant consumption.

An anlysis was made to determine the effect of an error in applying the velocity increment (ΔV) at the 50 n mi apoapsis. A 1 percent

[redacted]

[redacted]

error in ΔV would result in a periapsis altitude error of 2700 ft. There would also be periapsis velocity errors; however, from a propellant requirement standpoint, these would be compensated by the oppositely directed altitude errors.

The descent from 30,000 ft is accomplished using a constant thrust engine with the thrust vector always oriented opposite-to and colinear with the velocity vector. This trajectory results in low propellant consumption but is restricted in the sense that only one thrust-to-weight ratio can be used to bring the vehicle to zero velocity at a given altitude from the periapsis. For the 30,000 ft periapsis this thrust-to-weight ratio is 0.68 earth g. The ratio can be decreased by initiating thrust at points other than the periapsis. However, more propellant would be consumed by so doing, and it happens that, quite fortunately, the 0.68 ratio is approximately optimum from a propellant vs engine weight balance. The ideal velocity increment required for this maneuver is 5680 fps.

An altitude of 2600 ft at zero velocity results from the selected trajectory parameters given in the previous paragraphs. This altitude was selected to accommodate thrust vector errors of 0.5° and 1 percent in direction and magnitude respectively which could cause deviations of

[redacted]

as much as 2500 ft in the final altitude. Associated with this altitude error is also the possibility of incurring laterial deviations of up to 3000 ft which must be corrected during the final maneuver.

A maneuver employing a landing site beacon may be used to correct deviations from the desired landing trajectory during flight. This system would have the advantage of landing the vehicle directly on the target at zero velocity so that no additional hovering or lateral maneuver would be required and consequently a lower overall velocity increment would be needed.

A study of such a guided landing maneuver was begun, but results were not realized in the contract period. Based on the partial results obtained it is believed that a guided system with throttled engine could be devised to land with a velocity increment approximately 100 fps greater than that of the thrust-parallel-to-velocity maneuver with no hovering maneuver included, i.e., approximately 5980 fps ΔV.

As a result of the constant thrust deorbiting maneuver it was stated that the maneuver would terminate at an altitude of from 100 to 5100 ft and a lateral position error as great as 3000 ft. A throttlable propulsion system was selected to complete the terminal phase of the landing maneuver.

Considering an initial position of 5100 ft altitude and 3000 ft displacement, the vehicle would be allowed to fall for a short time to establish a 40 fps descent velocity. Alternately, the thrust could be terminated on the previous phase while 40 fps descent velocity still remained. The thrust level is then adjusted to provide a vertical thrust-to-weight ratio of 1.0 (lunar) so that a constant descent trajectory results. The descent time (t_b) is then approximately 128 sec. During this time the vehicle must first accelerate laterally towards the target and then decelerate to zero lateral velocity. The acceleration required to accomplish this is 0.139 lunar g which is directed towards the target for the first 64 sec and away from it for the last 64 sec. The net thrust-to-weight is then 1.01. The burnout thrust-to-weight from the previous maneuver is 6.27; this is the magnitude of the throttling ratio at the beginning of this maneuver. The ideal velocity requirement is then calculated to be 680 fps from

$$\Delta V = (F/W) g_o \, t_b$$

This results in a further throttling ratio of 1.06. The over all throttling ratio of 6.6 can be accomplished by actual throttling of a single engine, or alternately if the initial throttling is accomplished by shutting down some of the engines of a clustered system; the actual throttling requirement is then only about 6 percent.

A short burst of additional thrust is applied when nearing touchdown to reduce the velocity to a small value on contact. An additional allowance of 40 fps was made for this final braking thus bringing the total to 720 fps. The 40 fps descent velocity was selected as a compromise between getting down as quickly as possible to minimize propellant consumption and descending as slowly as possible to allow for translation and to minimize the possibility of vehicle damage if the final propulsive burst were not correctly applied. Various methods of impact shock abatement have been found in the literature (gas filled bags, frangible materials, etc.) but it appears that retro-rocket braking is most suitable for absorbing the large amounts of energy involved in this maneuver.

The alternate extreme situation in which the descent from orbit maneuver places the vehicle near the ground (100 ft altitude) but displaced from the target by 300 ft should be considered. The propellant requirements for this landing are not as high as the previous case because the burning time can be shorter and a higher thrust level can be used in the direction of translation. For example, if a 5 fps descent rate is maintained to ground contact, the horizontal acceleration will be 30 ft/sec^2 and the propulsion velocity requirement is 610 fps.

LUNAR TAKEOFF AND ORBIT ESTABLISHMENT

The maneuver possibilities investigated for the return voyage are shown in Fig. 3-114; the shaded areas represent the recommended maneuvers for this phase of the mission.

The results of the studies presented in the interim report show that the minimum propulsion velocity is required by a thrust parallel to velocity maneuver which places the vehicle directly in the transfer trajectory. The velocity requirements are shown in Fig. 2-124. The disadvantage of this maneuver is that a specific burnout angle is associated with each thrust-to-weight ratio and transfer time. Since the vehicle may have landed from orbit on any point on the lunar surface the above take off maneuver will, in general, not result in the proper orientation of the velocity vector at burnout. It should be mentioned that if the landing was accomplished directly without the use of an intermediate orbit then the direct takeoff will result in near correct velocity orientation at burnout and is applicable. In the general case the vehicle would rise vertically for a short time, rotate the velocity vector, and then direct thrust parallel to velocity until the flight conditions are such that the vehicle coasts to a 50 n mi apoapsis after the thrust is terminated. At the 50 n mi altitude the engine is reignited and a variable pitch angle program is initiated in which the thrust vector is oriented so as to maintain the altitude while increasing the circumferential velocity until

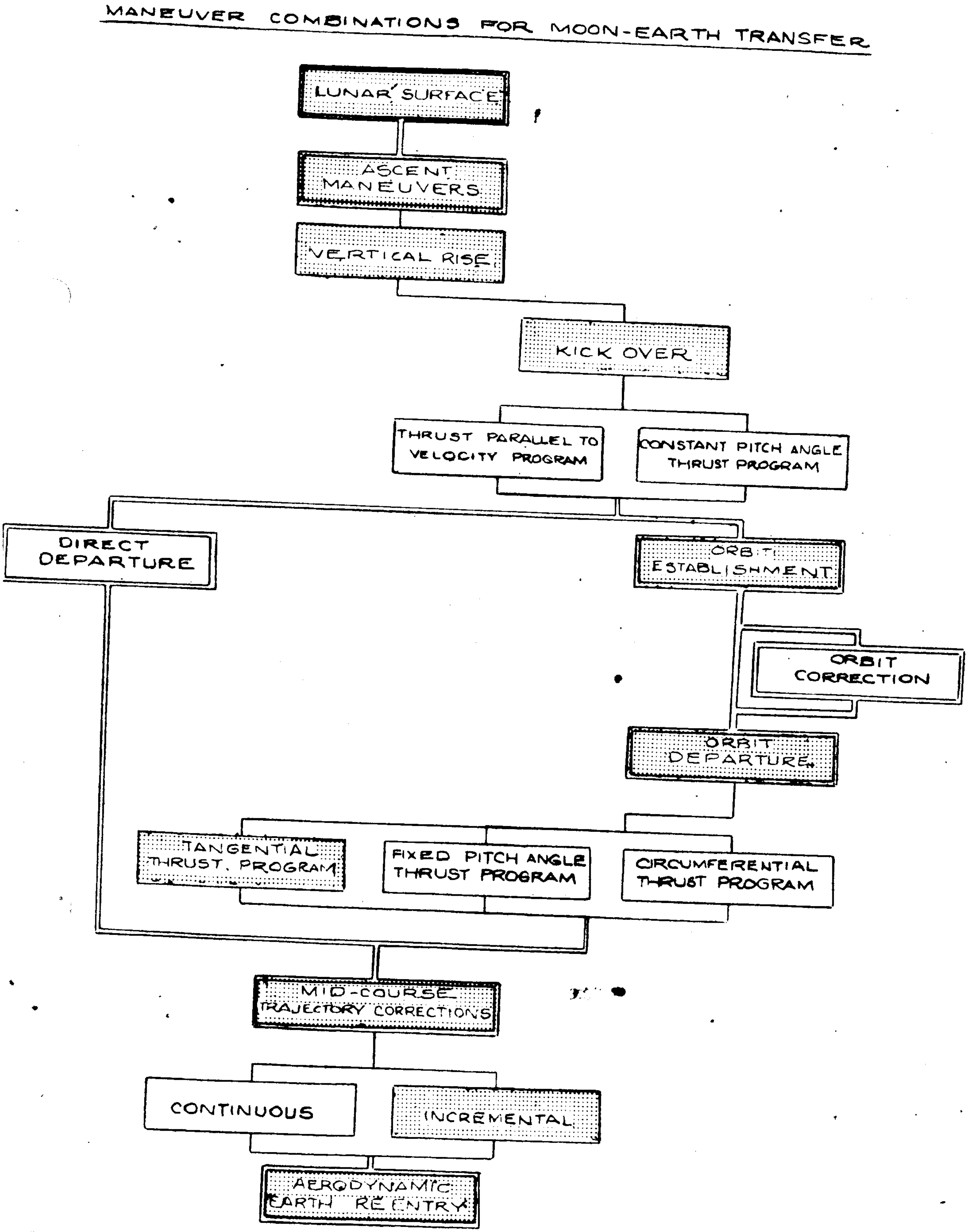


Figure 3-114. Maneuver Combinations for Moon-Earth Transfer

[redacted]

orbital velocity is reached. The first burnout conditions (magnitude and direction of the velocity vector and altitude) are such that if reignition at the 50 n mi altitude fails, the vehicle will be in an elliptical orbit which does not intercept the lunar surface. The velocity requirements for this maneuver are shown in Fig. 2-227. Errors in the orbit establishment are not critical because they can be compensated for when the injection into the Earth return transfer trajectory is accomplished.

RETURN TRANSFER, MIDCOURSE, AND RE-ENTRY

The 2.6 day return trip was selected for the same reasons mentioned in Section II. Propulsion velocity requirements are shown in Fig. 2-115 and 2-116. The maneuver is executed by aligning the thrust vector with the velocity vector until the desired hyperbolic velocity is obtained. This thrust program minimizes the propellant consumption requirements. An allowance of 150 fps was made for midcourse correction maneuver propulsion.

A study of the dynamics of re-entry using atmospheric braking is beyond the scope of this report. The factors involved in the optimization are the decelerative forces, the maximum heat transfer rate, the total heat transferred, and the velocity required to prevent skipping out of

[redacted]

the atmosphere. References 2 and 10 infer that aerodynamic re-entry is possible. Since adequate information was not available to determine the weight of the re-entry system this equipment must be considered as part of the payload weight in this report.

LUNAR VEHICLE STAGING

For the lunar and return mission a large number of propulsive maneuvers must be accomplished. The staging of the vehicle used to accomplish these maneuvers can be designed in a variety of ways depending upon the vehicle propellants, feed system, maneuver combinations, etc. It is the purpose of this study to survey a number of lunar vehicle designs and from this survey select the most desirable system(s). The desirability of a system design will depend upon the performance in terms of the delivered payload, system complexity, and the general attractiveness of the maneuver-staging combinations.

Two lunar landing and return maneuver methods have been selected for consideration: (1) direct lunar landing, and (2) landing from lunar orbit. The propulsive maneuvers of which these missions are comprised are listed in the maneuver key, Table 3-13. Both methods consider a mission initiated in an Earth satellite orbit, an Earth/Moon transfer, and mid-course trajectory corrections. The direct lunar landing proceeds to

[redacted]

contact the lunar surface directly from the transfer phase. The landing from lunar orbit first establishes a series of lunar orbits and then proceeds to contact the surface. Similar procedures are used in takeoff from the Moon and return to Earth.

This staging study considers the lunar mission from its initiation in an earth satellite orbit until the vehicle is on the Moon/Earth transfer back to Earth. The lunar landing and return mission is initiated from a 300 n mi earth orbit. The vehicle gross weight was 354,000 pounds; the payload which can be placed in a 300 n mi earth orbit by a NOVA H-6 booster. The determine vehicle staging for this mission, ten separate maneuvers were considered (Table 3-13). Nominal velocity increments are given to indicate the magnitude of each maneuver considered. The actual velocities used in the study varied slightly from this value depending upon the thrust-to-weight ratio.

The four propulsion systems discussed in another section of this report were considered for this study. These two propellant combinations (MON/MMH and LO_2/LH_2) and feed systems (pump and pressure-fed) represent a range of propulsion system variables sufficient to indicate any effects that may result.

[redacted]

TABLE 3-13

PROPULSIVE MANEUVERS

Designation	Maneuver	Nominal Velocity Increment, f
A	2.6 day Earth/Moon transfer	10,250
B	Two mid-course trajectory corrections	300
C	50 nmi lunar orbit establishment	3250
D	Low altitude elliptical orbit establishment	60
E	Descent from low altitude elliptic orbit	5800
F	Terminal maneuver; includes capability for hovering and translation	700
G	Takeoff to 50 nmi lunar orbit	5750
H	2.6 day Moon/Earth transfer from lunar orbit	3250
I	Direct landing from 2.6 day transfer	9200
J	Direct takeoff; thrust-parallel-to-velocity; 2.6 day Moon/Earth Transfer	9550

Where applicable the optimum thrust-to-weight ratio curve of Section were used to estimate stage thrust level. In other cases specific optimizations were conducted.

Staging was considered only at the beginning of maneuvers. The reason for this was that it was felt that the most critical phase of propulsion system operation is the startup. The orbital landing maneuvers were devised in such a manner that noncatastrophic propulsion system failure at the beginning of any maneuver would not result in the vehicle crashing. The same is true for the direct lunar landing and return mission (except for the actual landing phase), and the same staging philosophy was therefore employed for this mission.

Earth/Moon Transfer Phase

Maneuver A - 2.6 day Earth/Moon Transfer. The four propulsion systems discussed previously, plus a solid propellant propulsion system, were compared for the Earth/Moon transfer maneuver. The results of the vehicle comparison for this maneuver are shown in Fig. 3-115. These five vehicles are based upon a 354,000 lb gross weight in a 300 n mi earth orbit. The maneuver was not considered strenuous engough to require more than a single stage. The payload weights are shown and relative magnitude indicated.

For the liquid propellant systems the higher thrust levels of the pump fed systems were selected from the thrust optimization curve and are

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Initial Gross Weight in 300 n. mile Earth Orbit = 354,000 lbs.
(Based on NOVA H-6 Capability)

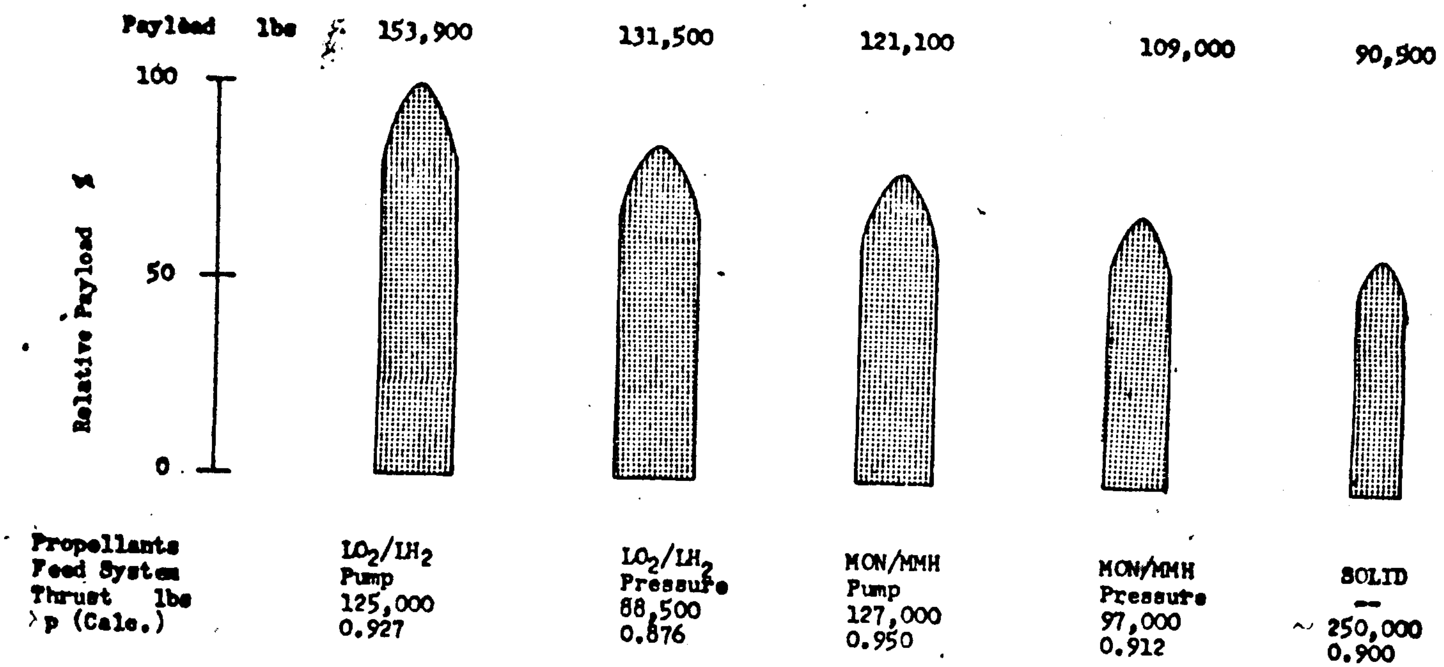


Figure 3-115. Earth/Moon Transfer Propulsion 2.6 Day Coast Time

[redacted]

dictated by the lower engine specific weights. The high thrust of the solid propellant system was considered necessary because of the solid propellant requirement for short burning times. The resultant propellant fractions (λ p) are indicative of the engine specific weight, the type of feed system and propellant, and the amount of propellant burned. These propellant fractions are somewhat higher than those that might be normally associated with these propulsion systems. This is because of the thrust-to-weight ratios of those space systems being considerably lower than those of comparable earth oeration systems. Because of the large payload advantage of the Liquid Oxygen/Liquid Hydrogen pump-fed propulsion system, it was selected as the only liquid propellant system to accomplish this first maneuver. For the short times that the propellants of this stage must be stored, no difficulties are anticipated with the Liquid Oxygen/Liquid Hydrogen combination.

<u>Maneuver B - Mid-course Trajectory Corrections</u>. The mid-course trajectory correction energy requirements as described in the maneuver section are very small as indicated by the table. This small propulsion requirement may be provided by briefly firing the main engine to avoid the complexity of a separate propulsion system. Two considerations must be made when using the main propulsion system: (1) propellant settling

prior to firing, and (2) the magnitude of the cutoff impulse. The former problem may be solved by firing a 1-lb-thrust settling engine for several minutes prior to main engine ignition. This engine may be fed from the same storable propellant source as the attitude control propulsion system. The relatively long settling time does not preclude the use the low thrust system because the time at which the midcourse correction is to be applied is known and the firing of the small engine can be programmed to lead the main engine firing by the appropriate amount.

Analysis of the effects of cutoff impulse deviations indicates that the anticipated deviations would generally result in uncertainties in velocity of less than 1 fps. Thus, the propellant settling engine could again be used as a vernier correction system.

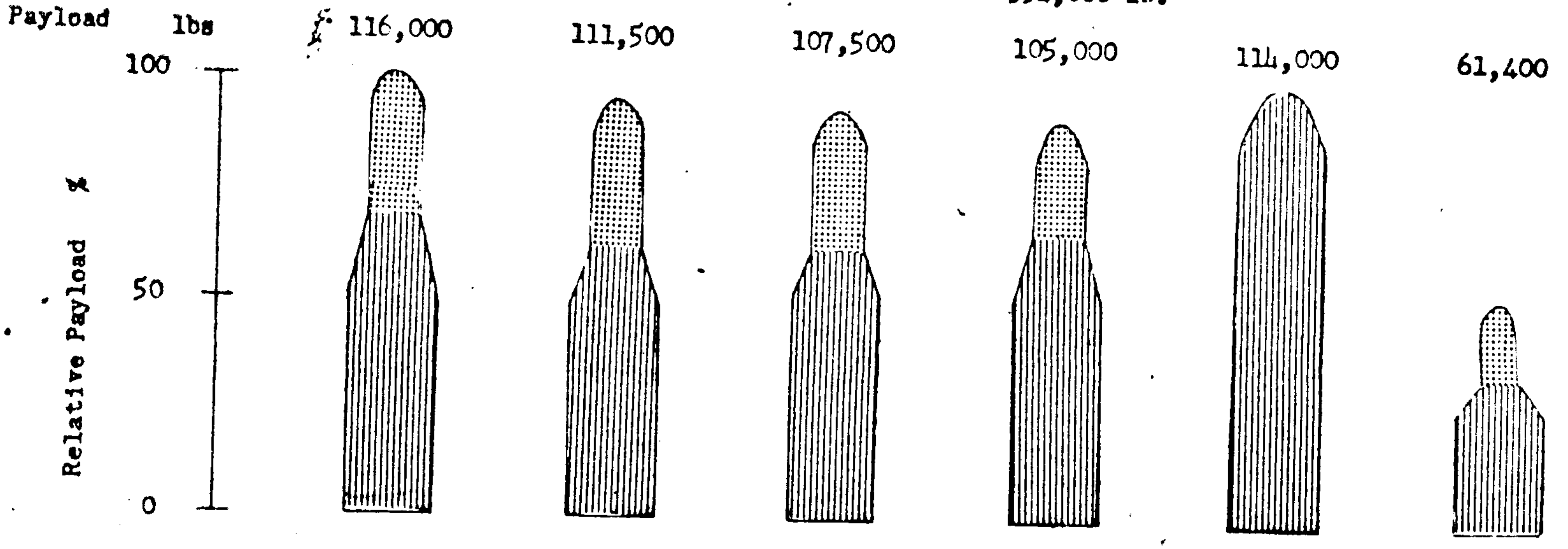
For purposes of this study a weight allowance is made for a propulsion system to accomplish the mid-course corrections. This weight allowance is based on the MON/MMH pressure-fed system.

Orbit Landing and Return Mission

Maneuver C - 50 n mi Lunar Orbit Establishment. The results of the vehicle comparison for this maneuver are shown in Fig. 3-116. Six vehicle configurations were compared for this maneuver. The first four systems

2.6 Day Earth/Moon Coast Time
Two Mid-Course Corrections
50 n. mile Lunar Orbit

Initial Gross Weight in 300 n. mile Earth Orbit = 354,000 lb.



2nd Stage						
Propellants	LO_2/LH_2	LO_2/LH_2	MON/MMH	MON/MMH		Solid
Feed System	Pump	Pressure	Pump	Pressure		
Thrust ~ lbs	29,000	20,000	30,000	22,000		
λ p(calc)	.906	.857	.936	.905		
1st Stage						
Propellants	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	Solid
Feed System	Pump	Pump	Pump	Pump	Pump	--

Figure 3-116. Lunar Orbit Establishment Propulsion

considered the Earth/Moon transfer maneuver (A) to have been performed by a Liquid Oxygen/Liquid Hydrogen pump-fed stage, and compares the (for this orbit establishment maneuver) four propulsion systems discussed previously. The other vehicles considered were a single-stage, Liquid Oxygen/Liquid Hydrogen pump-fed system that provides both the Earth/Moon transfer velocity and the lunar orbit establishment velocity increment, and a two-stage solid propellant vehicle.

The payloads in lunar orbit and the characteristics of the second stage are indicated. Thrust-to-weight ratios of the stages were determined from the optimization curves. The thrust available for the lunar orbit establishment in the single stage vehicle was dictated by that used in the Earth/Moon transfer.

All of the liquid propellant systems provide payloads of similar magnitude. The solid propellant system provides considerably less payload. The payload in lunar orbit indicates two systems worthy of further consideration. These are the Liquid Oxygen/Liquid Hydrogen pump-fed, two-stage and single-stage vehicles.

Maneuver D - Low Altitude Elliptical Orbit Establishment. This maneuver places the space vehicle on a 50 n. mi. x 30,000 ft elliptical orbit, a very efficient method of bringing the vehicle to a low altitude. The main propulsion system could be briefly fired again for this maneuver. Propellant settling would be accomplished by the same system employed for that purpose in the mid-course maneuvers. Here again, the deviation in velocity due to cutoff impulse uncertainties is less than 1 fps. The effect of this error on the periapsis is not severe and may be compensated for by the next maneuver.

<u>Maneuver E&F Lunar Descent and Landing</u>. Table 3-14 describes in detail the systems considered from earth orbit to lunar landing, and indicates the payload on the lunar surface for each of these systems. Seven combinations were considered ranging from four stages to a single stage. The propulsive maneuvers accomplished by each stage are indicated by the letter designation defined in the maneuver table (Table 3-13). The amount of throttling, both step and continuous, is indicated as well as the number of restarts.

This comparison is made to study the various staging methods of contacting the lunar surface. Maneuvers E and F as described in the table are used to achieve contact. Inherent in maneuver F is a hovering phase of operation which requires that a thrust-to-weight ratio of 1.0 (lunar) be maintained. This phase of operation dictates the throttling requirements. In the combinations where maneuver F was performed by a separate system, from the one accomplishing maneuver E, a parallel system was considered. The parallel system means that there are two propulsion systems

TABLE 3-14

LUNAR LANDING PROPULSION

	I	II	III	IV	V	VI	VII
Payload on Lunar Surface lb	66,500	60,500	61.800	65,200	59,200	66,500	59,200
4th Stage							
Propellants	LO_2/LH_2	MON/MMH					
Feed System	Pump	Pressure					
* Maneuvers	F	F					
Throttling	Continuous 6 percent	Continuous 8 percent					
Restarts	0	0					
Comments	Parallel System	Parallel System					
3rd Stage							
Propellants	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	MON/MMH	MON/MMH		
Feed System	Pump	Pressure	Pressure	Pressure	Pressure		
* Maneuvers	D, E	D, E	D, E, F	F	F		
Throttling	--	--	Step 6:1 Continuous 6 percent	Continuous 8 percent	Continuous 8 percent		
Restart	1	1	1	0	0		
Comments				Parallel System	Parallel System		
2nd Stage							
Propellants	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	
Feed System	Pump	Pump	Pump	Pump	Pressure	Pump	
* Maneuvers	C	C	C	C, D, E	D, E	D, E, F	
Throttling	—	—	--	--	—	Step 6:1 Continuous 6 percent	
Restarts	0	0	0	2	1	1	
Comments				Restart in orbit		No System Ignition During Descent	
1st Stage							
Propellants	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2
Feed System	Pump	Pump	Pump	Pump	Pump	Pump	Pump
* Maneuvers	A, B	A, B	A, B	A, B	A, B, C	A,B,C	A, B,C,D, E,F
Throttling	--	—	—	—	—	—	Step 8:1 Continuous 8 percent
Restarts	2	2	2	2	3	3	5
Comments							

* See maneuver key in text

[redacted]

in the same stage. The parallel system was considered instead of vehicle staging during the terminal maneuver because of the inefficiency of staging for such a small propulsive maneuver and the difficulty of accomplishing the staging in this type of maneuver. This terminal landing system would be ignited before shutdown of the larger system so that there would be no period of free fall during this terminal maneuver.

An alternative to using a parallel system would be step throttling the system used to cancel the orbital velocity (Maneuver E). This system would require step throttling by as much as 8:1. Step throttling can be accomplished by either a single engine or a cluster of engines as described in the discussion of throttling. No difficulty is anticipated with either method of operation. The redundancy and, therefore, reliability possibilities available with the clustered engine concept may render it the more desirable method.

From the comparison of payload capability three systems appear interesting: Systems I, IV, and VI. System I uses a separate Liquid Oxygen/Liquid Hydrogen, pump-fed storage for each major propulsive maneuver. This is not considered further because the small payload gain does not appear to warrant the increased number of stages. The other two systems both appear attractive and will be considered further.

Maneuvers G and H - Takeoff and Return to Earth. For the lunar takeoff and return to Earth a combination of maneuvers G and H are generally considered. This use of a lunar orbit allows a greater variety of lunar takeoff sites as discussed previously . The maneuver (G) used for orbit establishment uses a trajectory with an intermediate coast phase. This maneuver requires a restart after the coast phase to initiate the portion of the maneuver that establishes the orbit. Although this restart is not critical, because the first phase of the maneuver establishes the vehicle in an elliptical orbit, it may be eliminated through use of a powered-all-the-way trajectory. As a comparison Maneuver J, a direct takeoff is included.

Seven staging combinations were studied; the results are presented in Table 3-15. The first six systems in the table use the first stage to accomplish the Earth/Moon transfer alone. The seventh system uses the first to accomplish both the transfer and orbit establishment maneuvers (A, C).

Of all of the operational phases of the lunar landing and return mission, the return maneuver is the phase in which propellant storage problems are most likely to be encountered. The moderately long storage times (a week to 10 days) necessary, and the fact that this is the smallest stage of the entire vehicle may create difficulties in maintaining cryogenic propellants. Figure 3-34 indicates that for the vehicle sizes

TABLE 3-[illegible]

Moon/Earth Transfer Propulsion

2.6 Day Coast Time

Note: With the exception of VII, all vehicles considered use (1) a LO_2/LH_2, pump-fed system to accomplish Earth/Moon transfer (A) and (2) a parallel MON/MMH, pressure-fed system for Lunar terminal maneuver (F)

	I	II	III	IV	V	VI	VII
Payload* on Moon/ Earth Transfer lbs	26,100	22,400	26,300	27,800	16,200	26,400	29,500
3rd Stage							
Propellants	LO_2/LH_2	MON/MMH	LO_2/LH_2			MON/MMH	
Feed System	Pressure	Pressure	Pressure			Pressure	
** Maneuvers	G,H	G,H	J			H	
Restarts	2	2	0			0	
Comments	2nd Restart in Orbit	2nd Restart in Orbit	Thrust Parallel to Velocity				
2nd Stage							
Propellants	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2
Feed System	Pump	Pump	Pump	Pump	Pressure	Pump	Pump
** Maneuvers	C, D, E	C, D, E	C, D, E	C, D, E, G, H	C, D, E, G, H	C, D, E, G	D, E, F, G, H
Restarts	2	2	2	5	5	4	4
Comments	Restart in Orbit	Restart in Orbit	Restart in Orbit	2 Restarts in Orbit; 1 Restart on Lunar Surface	2 Restarts in Orbit; 1 Restart on Lunar Surface	1 Restart in Orbit; 1 Restart on Lunar Surface	1 Restart in Orbit; 1 Restart on Surface

* Includes Weight of Mid-Course Correction System for Moon/Earth Transfer

** See maneuver key in text

[redacted]

being considered here the cryogenic combinations could be used with no difficulties. A reduction in the size of the over-all space vehicle may, however, cause the storable propellant systems to be more attractive.

Keeping these considerations in mind Systems II and V were eliminated from the payload capability standpoint. Systems I and III are essentially the same with the maneuver combinations of System I offering more flexibility. From consideration of staging and the payload capability System VII is felt to be the most desirable.

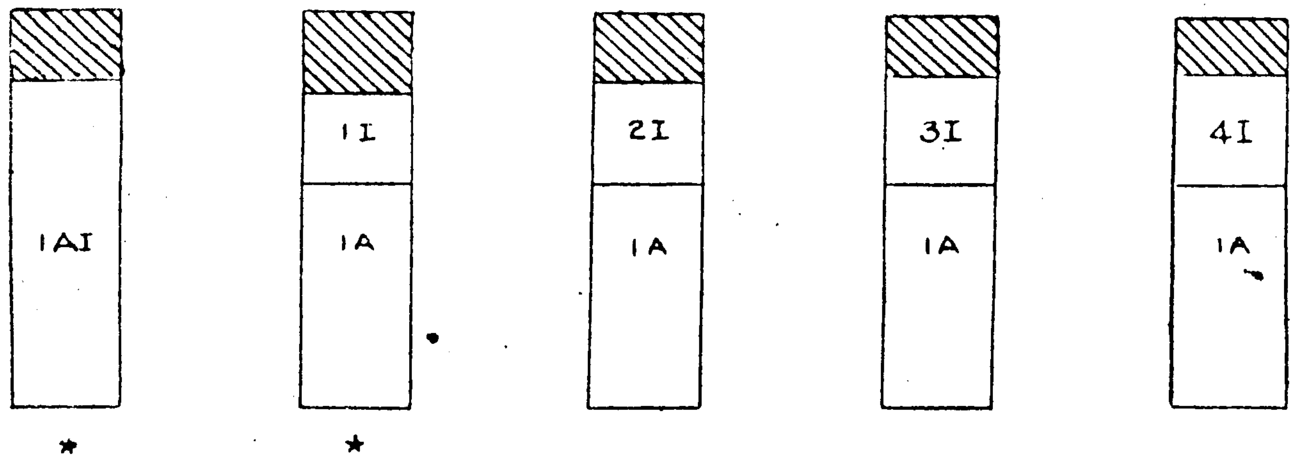
Direct Landing and Return Mission

The analysis of the 2.6 day transfer from a 300 n mi earth orbit to the moon performed in the previous section is applicable to both direct and orbital lunar landings. Superiority of the pump-fed Liquid Oxygen/Liquid Hydrogen system was found as shown in Fig. 3-116. The payload capability of this stage was 151,500 lb when the mid-course correction propulsion requirement was included.

Maneuver I - Direct Lunar Landing. Having selected the liquid Oxygen/liquid Hydrogen pump-fed system for the first propulsion phase, the next step was to consider the effects of (1) reigniting this stage for the direct lunar landing and (2) starting a second stage for this maneuver. The results are shown in Fig. 3-117.

When staging took place after the transfer maneuver, the payload landed on the moon by the second stage depended on the propulsion system selected for that stage. Here again the liquid oxygen/liquid hydrogen pump-fed system stood out in payload capability and was selected for further study. Since no storage problem was anticipated for vehicles of this size, the MON/MMH systems were eliminated as well as the pressure-fed liquid oxygen/liquid hydrogen landing stages on the basis of payload. The single-stage liquid oxygen/liquid hydrogen pump-fed system was also retained for its simplicity, notwithstanding the 10 percent payload shortcoming of this system. When considering the single stage for both maneuvers (transfer and landing) a different optimum thrust-to-weight was obtained than for the transfer maneuver alone. This was because of the fact that the initial thrust-to-weight for the landing maneuver depends on the initial thrust-to-weight for the transfer maneuver. Since the optimum thrust-to-weight for the landing was higher than that resulting from the optimum thrust-to-weight of the transfer, the initial thrust-to-weight for the dual maneuver was therefore greater than that of the transfer. The optimum thrust-to-weight was a compromise of the requirements of the individual maneuvers.

[redacted]



STAGES	IAI	IA	1I	IA	2I	IA	3I	IA	4I
STAGE WT. (K)	292	202.5	82.5	202.5	93	202.5	95	202.5	101.5
F/W_{OPT} (EARTH)	0.70	0.35	2.00	0.35	1.50	0.35	2.10	0.35	1.60
PAYLOAD (K)	62	69.0		58.5		56.5		50.0	

STAGE CHARACTERISTICS
1. LO_2/LH_2 PUMP FED
2. LO_2/LH_2 PRESS. FED
3. MON/MMH PUMP FED
4. MON/MMH PRESS. FED

A EARTH-MOON TRANSFER

I DIRECT LUNAR LANDING

★ SELECTED FOR FURTHER STUDY

Figure 3-117. Payload on Moon by Direct Landing

[redacted]

The pressure-fed systems exhibited lower optimum thrust-to-weight ratios than the pump-fed systems primarily because of the heavy engines employed by the pressure-fed configurations. Thus, although increasing the thrust level tended to reduce the propellant and tank weights the engine weights soon overrode this advantage.

There was a rather broad region near the optimum thrust-to-weight ratio where the payload variation with thrust-to-weight was quite small, as shown in Fig.3-118, which was drawn for the selected liquid oxygen/liquid hydrogen pump-fed second stage. The other stages considered exhibited an analogous behavior in this respect.

The unusually high thrust-to-weight ratios at which the second stages optimized are due mainly to the shape of the velocity increment, ΔV, vs thrust-to-weight curve for the landing maneuver. A comparison of the curves for vertical landing with a thrust-antiparallel-to-velocity maneuver is shown in Fig. 2-122. The generally lower velocity increment and the more pronounced "knee" of the latter curve permit operation at lower values of thrust-to-weight before prohibitive velocity requirements are encountered. The high ΔV of the direct descent maneuver is due to the high gravity losses associated with the vertical velocity vector during firing.

[redacted]

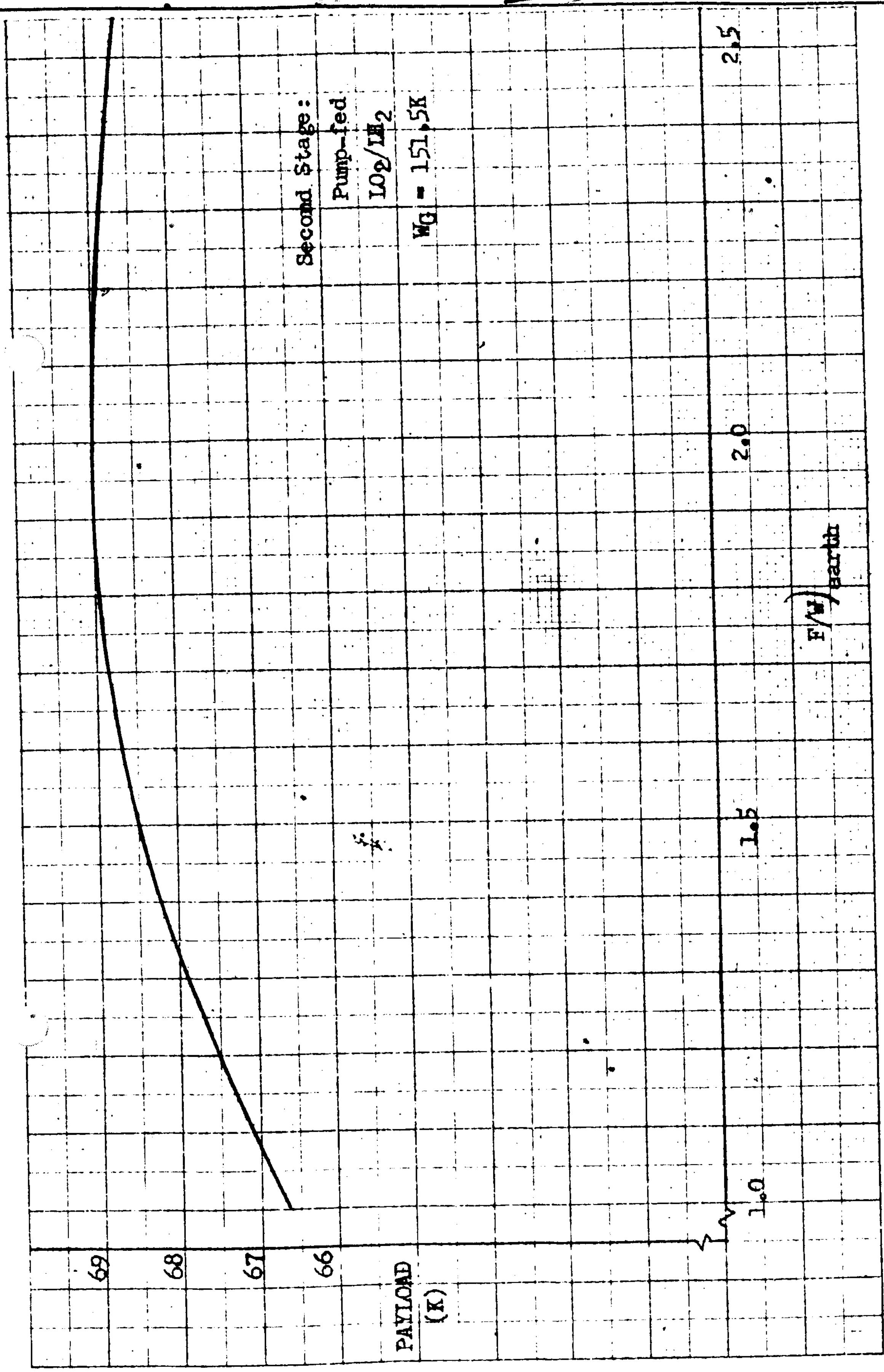


Figure 3-118. Payload on Moon vs Second Stage Thrust/Weight

[redacted]

The relative shapes and positions of these curves also explain why the payload capability of the vehicle using the direct landing maneuver is less than that of the vehicle employing the orbital landing technique. This is because the curve for the orbital landing more closely resembles that of the thrust-parallel-to-velocity maneuver than the curve for the vertical landing.

Maneuver J - Direct Takeoff and Return to Earth. The staging for the return trip was next investigated. Using the gross weight available from the two-stage transfer-landing vehicle, several feed system and propellant combinations were considered for the return stage. Results of these analyses are presented in Fig. 3-119 and show the payload advantage of the liquid oxygen/liquid hydrogen pump-fed system. Substitution of the pump-fed, storable third stage resulted in a 20 percent payload loss.

Two-stage vehicles were also considered for the total mission. To minimize payload losses, the first stage of these vehicles was selected as the liquid oxygen/liquid hydrogen pump-fed system. A similar system was used as the second stage and two methods of staging were studied: (1) staging after transfer, and (2) staging after lunar landing. The former resulted in a payload of 26,800 lb and the latter 26,300 lb. A storable second stage was then substituted in the latter case and the payload available for earth re-entry dropped to 19,000 lb.

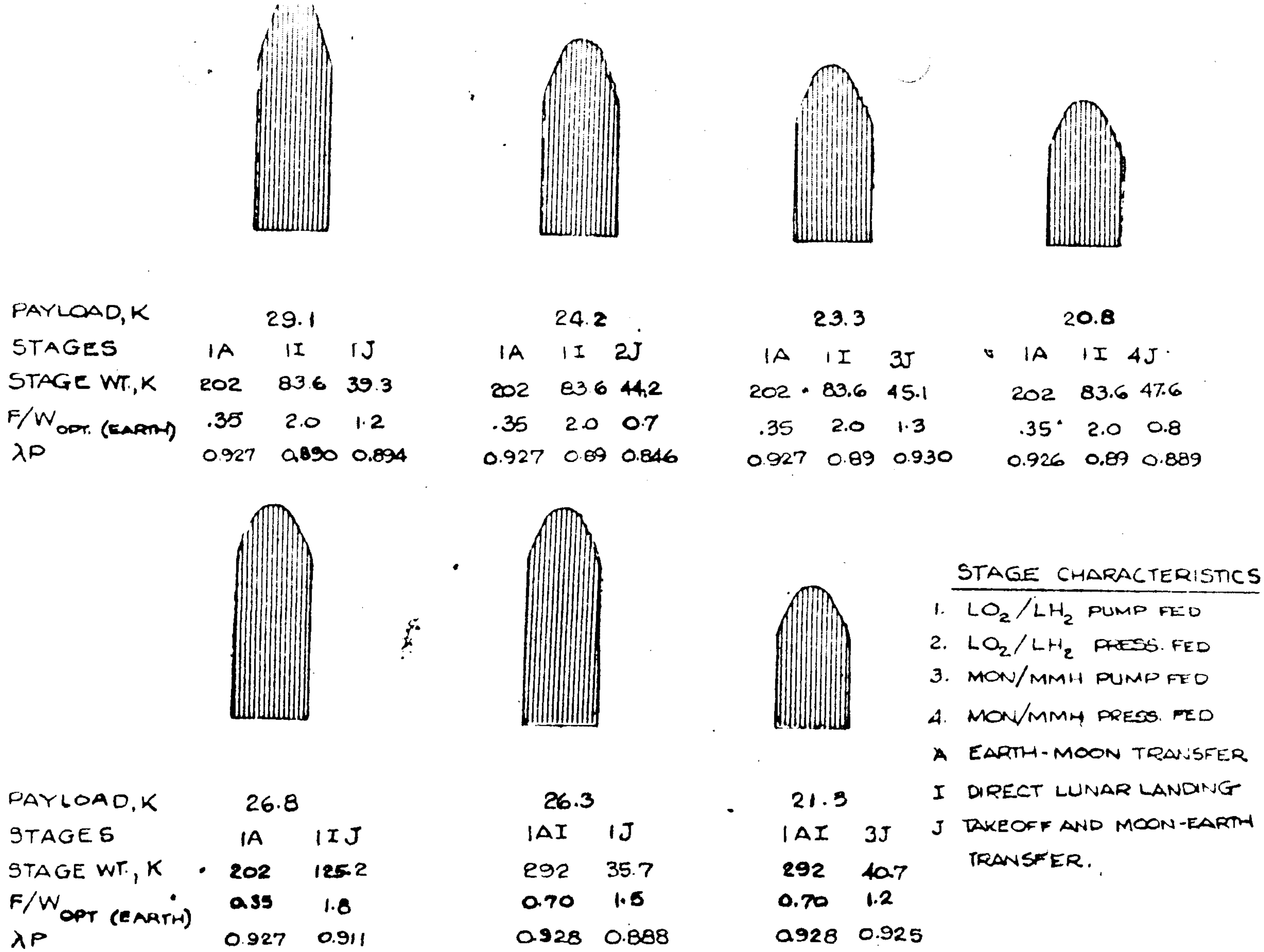


Figure 3-119. Earth Re-entry Payloads for Direct Lunar Landing and Return

Despite a 10 percent payload disadvantage the two-stage pump-fed liquid oxygen/liquid hydrogen vehicle was selected in preference to the three stage configuration. The reason for this choice was the inherent simplicity and lower cost of the two-stage system. Staging after landing was preferred because this arrangement permits the landing to be accomplished by the already proven transfer propulsion system, and also because any damage to the landing engine at touchdown would not affect the return propulsion system.

The study of space environment indicated that storage of the cryogenic propellants on the moon is not problematical enough to warrant substitution of storable propellants for the return stage. This section shows that, depending on the amount of thermal conduction by the structure, storage times of from 1 month to 3 years are possible. The former represents a very pessimistic conduction estimate and the latter represents the situation of no structural conduction. Figure 3-34 represents a conduction estimate of 10 btu/hr which results in an allowable storage time of approximately 8 months.

The variation of payload with second stage thrust-to-weight of the selected system is shown in Fig. 3-120. The extremely broad peak of this curve indicates that the thrust level could be varied from 35,000 to 80,000-lb thrust with no significant loss in payload.

[redacted]

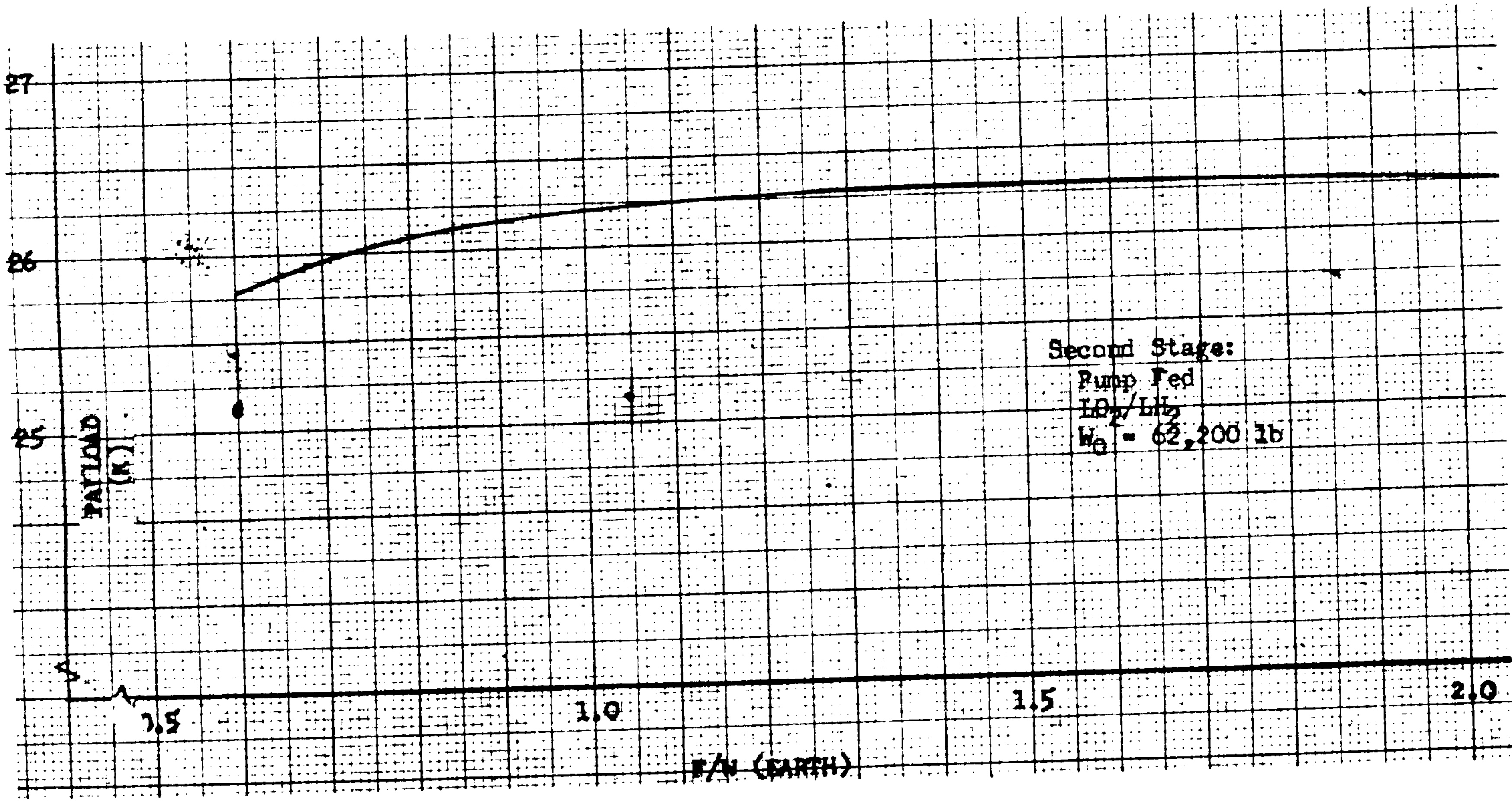


Figure 3-120. Payload Available for Earth Re-entry vs Second Stage Thrust/Weight Ratio

[redacted]

[redacted]

RECOMMENDATIONS FOR LUNAR LANDING AND RETURN MISSION USING AN INTERMEDIATE LUNAR ORBIT

Applicability

Lunar landing and return missions using intermediate lunar orbits are suggested for use with manned payloads. Maneuvers used in the vicinity of the lunar surface provide a systematic, reliable method of landing. This step-by-step approach is compatable with existing guidance and propulsion system accuracies. The use of the intermediate orbits not only provides ample opportunity for landing selection but provides mission flexibility and reliability for manned missions.

Maneuvers

The space vehicle will provide the propulsion to leave the earth orbit and achieve the velocity required for the Earth/Moon transfer; a thrust parallel to velocity maneuver can be employed. A series of mid-course trajectory corrections are applied during the transfer.

In the vicinity of the Moon thrust is applied anti-parallel to velocity to establish a lunar orbit (50 n mi). After observing the Moon for some period of time and selecting a tentative landing site, a small velocity increment is supplied to establish a 30,000 ft by 50 n mi elliptical orbit. From the periapsis of this orbit a thrust anti-parallel to velocity

[redacted]

maneuver is used to arrive at a point close to the lunar surface. The altitude of this point is governed by the guidance and propulsion system accuracies. From this point a constant velocity descent results in contact with the lunar surface. Included in this latter phase is the capability of a 3000 ft translation. The takeoff and return to earth maneuver includes the establishment of a lunar orbit before beginning the return transfer.

Vehicle

The space vehicle recommended for this method of accomplishing the lunar landing and return is described in Table 3-16. This space vehicle can be placed into a 300 n mi earth satellite orbit by a NOVA H-6 booster vehicle. It will perform a soft landing on the lunar surface, takeoff, and place a 29,500 lb payload on a coast trajectory to earth. This 29,500 lb includes the weight of trajectory correction systems for the return trip and any re-entry systems required.

This space vehicle consists of two stages and main propulsion systems which are pump-fed using the liquid oxygen/liquid hydrogen propellant

TABLE 3-16

LUNAR LANDING AND RETURN VEHICLE USING AN INTERMEDIATE LUNAR ORBIT

Payload Weight on Moon/Earth Transfer, lb	29,500
Stage Two	
Propellants	LO_2/LH_2
Feed System	Pump
Throttling	6:1 Step 6 Percent Continuous
Restarts	4
Gross Weight, lb	114,000
Thrust, lb	91,000
Number of Engines (1 Redundant; 1 Throttleable)	7
Stage One	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	3
Gross Weight, lb	554,000
Thrust, lb	125,000
Number of Engines	1

[redacted]

combination. Auxiliary propulsion systems are used to accomplish attitude control maneuvers.

The first stage of the vehicle which is initiated in Earth orbit uses a single engine producing 125,000-lb-thrust. This stage provides propulsion for the Earth/Moon transfer, is shut down, and then reignited to be used in establishing the 50 n mi circular lunar oribt. Cutoff impulse deviations do not appear to prohibit using the main engine for midcourse correction.

The second stage is briefly used to change this orbit to a 50 n mi x 30,000 ft elliptical orbit, and is then reignited to provide thrust for landing. This second stage uses a seven engine cluster which produces a total thrust of 91,000 lb. One of the engines is designed to be redundant. At the end of the first phase of this descent from orbit all but one of the engines are shut down to provide the step throttling necessary for hovering or constant velocity descent. This remaining engine is capable of being continuously throttled by about 6 percent. The clustered engine technique appears advantageous over the use of a single engine with 8:1 throttleability ratio. The duration of this hovering maneuver depends upon the altitude at the beginning of the hover phase and the amount of lateral movement desired. These are in turn determined by the accuracy of the guidance. This terminal phase is ended and the last engine shut down when the vehicle contacts the lunar surface.

For takeoff from the lunar surface, the engine cluster is reignited and the vehicle enters a low-altitude lunar obrit. The engine is again restarted to leave the orbit for the return to Earth.

[redacted]

RECOMMENDATIONS FOR DIRECT LUNAR LANDING

Applicability

Direct landing is recommended for vehicles having simple terminal guidance systems which sense only altitude and descent rate during the actual landing phase, and having constant thrust propulsion systems. The mission is characterized by restricted landing area capability and CEP values of a few miles.

Maneuvers

The vehicle may leave the assumed 300 n mi earth orbit and achieve velocity to complete the transfer in 2.6 days. One or more mid-course corrections are applied during the transfer as determined by the guidance system and accuracy requirements.

At a predetermined point above the lunar surface, thrust is applied radially to reduce the descent velocity to a small value; thrust is then terminated and the vehicle falls freely (except for attitude control). If the first burnout altitude is too great from impact velocity considerations, a second firing is made which is followed by free fall to the lunar surface.

[redacted]

The return trajectory consists of a vertical rise until 2.6 day transfer velocity is attained followed by midcourse corrections to obtain the desired earth re-entry corridor. Aerodynamic re-entry is assumed.

Vehicle

A two-stage vehicle is recommended for the mission. Both stages are propelled by pump-fed liquod oxygen/liquid hydrogen propulsion systems. The recommended configuration is illustrated in Table 3-17. Staging is accomplished on the lunar surface with the following advantages: 1) the propulsion system used for the landing maneuver is the same one that has preven operable in the transfer maneuver, and 2) if boattail damage is incurred upon touchdown the return propulsion system will probably not be affected.

Attitude control systems are similar to those employed in the orbital landing lunar mission, however, no abort system is needed for this mission.

[redacted]

TABLE 3-17

RECOMMENDED SYSTEM FOR DIRECT LUNAR

LANDING AND RETURN MISSION

Payload Available for Earth Re-entry, lb	26,300
Stage Two	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	None
Gross Weight, lb	62,000
Thrust, lb	56,000
Stage One	
Propellants	LO_2/LH_2
Feed System	Pump
Restarts	4
Gross Weight, lb	354,000
Thrust, lb	248,000

AUXILIARY PROPULSION SYSTEMS

Abort Propulsion

The propulsion requirements for suborbital and superorbital aborts can be satisfied by the main propulsion system as stated in the mission analysis section. A system for high dynamic pressure and liftoff aborts is also described in that section. A cluster of four solid propellant rockets, each having a thrust-to-weight ratio of four (based on payload capsule weight), and a burning time of 5 sec is recommended. These are to be fired either two at a time or simultaneously depending upon the dynamic pressure level at abort. For a successful launch with no early abort requirements the solid rocket propulsion system may be jettisoned after leaving the high dynamic pressure region.

Attitude Control

The attitude control requirements may be divided into two areas: (1) when the main propulsion system is firing, and (2) during coast phase. The main propulsion system should include the capability of gimbaling the thrust chamber to provide pitch and yaw attitude control. The magnitude of the gimbaling requirement is dependent upon the dynamics of the missile. Roll-attitude control is required during this phase not only to position the vehicle, but also to counteract the torques caused by main engine thrust vector misalignment during gimbaling. Two 25-lb-thrust engines would be adequate for this purpose. The engines would be hinged and mounted on the circumference of the final stage. The propellants would be supplied by the same pressure source (pump-or pressure-feed) as the main engines. Reference 3 pointed out the desirability of maintaining continuous three-axis attitude control about a 5 deg dead band to eliminate a complicated search procedure to establish vehicle orientation later in the mission. The reference concluded that this can be done with a pressure-fed, storable-propellant, fixed-thrust (0.5 lb per engine) system weighing approximately 50 lb. When the propellant is scaled to the longer duration of the 2.6 day transfer the weight becomes approximately 70 lb. This includes propellants, plumbing, and six engines.

EFFECTS OF BOOSTER SIZE

The purpose of this study area was to determine the effects of using different boosters upon the philosophy employed in the design of the lunar space vehicle. It was recognized that the payloads of the differently staged vehicles would not shift drastically relative

[redacted]

to each other when the initial vehicle weight was changed. The method of analysis then was to assume the same velocity increment per stage as in the nominal vehicle and and to assume temporarily that liquid oxygen/liquid hydrogen propulsion systems were used. The payload and second stage gross weight thus derived were used as the basis for conclusions concerning the type of mission, propellant, combination, and feed system to be recommended for the vehicles which would be the upper stages of the various boosters considered. These results are summarized in Table 3-18 .

As the gross weight of a stage decreases the relative weight of the tank insulation increases as demonstrated in the section of this report concerning insulation weights. Figure 3-34 of that section shows the storage time beyond which the weight of the insulation for the liquid oxygen/liquid hydrogen becomes prohibitive when compared on the basis of payload capabilities with a storable propellant system of similar propellant weight. The relatively pessimistic assumption of 100 Btu/hr structural conduction is applicable to this graph. Extrapolation of this figure indicates that even for the smallest booster system considered (the C-1) a stopover time on the Moon of about 60 hr is allowable before the insulation weight dictates that storable propellants be considered for the second stage.

[redacted]

TABLE 3-18

PAYLOAD CAPABILITIES OF TWO-STAGE

LO_2/LH_2 PUMP-FED SPACE VEHICLES USING

VARIOUS BOOSTERS

BOOSTER VEHICLE	C-1	H-2	H-6 (Nominal)	H-8
GROSS WEIGHT IN ORBIT, LB.	24,300	118,000	354,000	472,000
SECOND STAGE PROPELLANT WEIGHT, LB.	4800	25,000	77,000	103,500
PAYLOAD RETURNED TO EARTH ATMOSPHERE, LB.	1300	9300	29,500	41,000

[redacted]

Another effect of booster selection is the type of mission to be flown. If it is assumed that the manned systems require a returned payload of approximately 10,000 lb it is seen that the H-2 booster is marginally inadequate and the C-1 is certainly useless for this mission. These vehicles would then fly the unmanned missions.

In trips to other planets or the moon, life on these bodies must be protected from any Earth species which might prove toxic. Where surface contact is possible the space vehicle must be sterlizied. Precautions must also be taken to prevent contamination in event of system failure. Where the propulsion system restarts it must in general be purged after shutdown of the previous firing.

In any propulsion system a number of tradeoffs must be considered in the system design. These will be facilitated by the use of exchange factors or influence coefficients which relate a change in some parameter to its effect on the system. Alternate methods and missions may be contemplated, and should be indicated.

TABLE 3-19

GENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

  A. Total Impulse Required (or Ideal Velocity Increment)

    1. Maximum; Mission

    2. Minimum; Mission

  B. Maximum Impulse (Velocity) Increment; Mission

  C. Minimum Impulse (Velocity) Increment; Mission

  D. Number of Increments

II. Thrust

  A. Magnitude

    1. Steady-State Design Thrust Magnitude

      a. Thrust-to-Earth Weight Ratio

      b. Absolute Value

    2. Tolerance

      a. Engine-to-Engine

      b. Run-to-Run

    3. Throttling

      a. Step

      b. Continuous

    4. Accuracy of Thrust Programming

    5. Number of Restarts

    6. Type of Thrust Control

TABLE 3-19

(Continued)

B. Transients

1. Start Sequence (Ignition and Response Time)

2. Startup Impulse

a. Nominal

b. Tolerance

3. Cutoff Impulse

a. Nominal

b. Tolerance

4. Throttling Transition and Response Time

a. Step

b. Continuous

C. Thrust Vector Control

1. Vector Control Requirement

2. Method of Control

3. Engine Thrust

a. Angular

b. Lateral

III. Propellants

A. Composition

B. Mixture Ratio

1. Nominal

2. Tolerance

3. Mixture Ratio Range

TABLE 3-19

(Continued)

C. Specific Impulse

1. Reference Engine Parameters

a. Mixture Ratio

b. Chamber Pressure

c. Expansion Ratio

2. Nominal Specific Impulse at Reference Conditions

3. Minimum at Reference Conditions

a. Run-to-Run

b. Engine-to-Engine

D. Compatibility with Manned Missions

E. Contamination Effects on Alien Environment

F. Temperature Effects

1. Density

2. Vapor Pressure

3. Heat of Vaporization

4. Heat of Fusion

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement

2. Number of Zero Gravity Engine Starts

3. Separation Method

4. Tank Venting

a. Requirement

b. Method

TABLE 3-19

(Continued)

B. Space Storage of Propellants

1. Environment

a. Thermal Radiation

b. Internal Heat Source

c. Ionizing Radiation

d. Meteoroids

2. Storage Time

3. Propellant Temperature Limits

4. Storage Methods

a. Exposed Surface Characteristics

b. External Insulation

c. Internal Design

d. Propellant Boil-off

e. Meteoroid Shield

f. Deleterious Effects of Environment on Storage Methods

g. Attitude and Geometry Limits

C. Component Design Restrictions

1. Meteoroids, Puncture

2. Temperature

3. Ionizing Radiation, Materials

4. Vacuum-materials, Start

D. Launch Environment

1. Thermal

2. Handling

[redacted]

TABLE 3-19

(Continued)

E. Target or Payload Contamination

1. External Contamination Sources

a. Bacteriological (Living)

b. Chemical (Non-living)

2. Decontamination Methods

a. Cleaning

b. Sterilization

3. Contamination in Event of System Failure

F. System Purging Requirements

1. Number of Purges

2. Type of Gas

3. Sequence

V. System Reliability as a Function of Development Time

A. Component

B. Engine

C. Vehicle

VI. Off Design Operation

A. Exchange Factors for Perturbation from Nominal

1. Engine Operating Parameters

a. Mixture Ratio

b. Chamber Pressure

c. Expansion Ratio

d. Thrust

2. Hardware Weight Equivalent of Specific Impulse

B. Alternate Mission Performance

[redacted]

TABLE 3-20

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants

b. Nominal Mixture Ratio

c. Propellant Temperature Limits

2. Useable Propellant

a. Maximum

b. Minimum

3. Reserve Propellant Weights

a. Flight Performance

b. Residual

c. Trapped

d. Fuel Bias

e. Boil-off Reserve

B. Tank Loads

1. Handling

2. Launch

3. In Atmosphere Flight

4. Space Flight

TABLE 3-20

(Continued)

C. Tank Pressure and NPSH

  1. Nominal

  2. Tolerance

D. Thermal Control

  1. Thermal Loads

    a. Ground Loads

    b. Aerodynamic

    c. Internal

    d. Space Loads

  2. Temperature Limits

E. Zero Gravity Requirements

  1. Gas/Liquid Separation Requirements

  2. Tank Venting Requirement

F. Staging, Geometry, and Configuration Requirements

G. Secondary Auxiliary Systems

H. Propellant Utilization System Requirements

I. Tank and Structure Weight Limits

7. Pressurization System

A. Purposes of Pressurization

B. Gas Volume in Propellant Tank

  1. Total

  2. Increments

TABLE 3-20

(Continued)

C. Gas Pressure in Propellant Tank

1. Nominal

2. Tolerance

D. Propellant Properties

1. Thermodynamic

2. Compatibility

E. Environment

1. Storage

a. Time

b. Gas Volume During Storage

2. Thermal Environment

3. Zero Gravity

F. Weight

III. Engine System

A. Propellant Description

1. Propellants

2. Thermodynamic Properties

3. Mixture Ratio

a. Nominal

b. Tolerance

B. Thrust

1. Nominal

2. Tolerance

3. Transients

[redacted]

TABLE 3-20

(Continued)

C. Type of Feed System

D. Specific Impulse

1. Nominal

2. Minimum

E. Engine Inlet Conditions

1. Storage

a. Time

b. Gas Volume During Storage

2. Thermal Environment

3. Zero Gravity

F. Weight

III. Engine System

A. Propellant Description

1. Propellants

2. Thermodynamic Properties

3. Mixture Ratio

a. Nominal

b. Tolerance

B. Thrust

1. Nominal

2. Tolerance

3. Transients

C. Type of Feed System

[redacted]

[redacted]

TABLE 3-20

(Continued)

D. Specific Impulse

1. Nominal

2. Minimum

E. Engine Inlet Conditions

F. Envelope Requirements

G. Throttling Requirements

1. Step

2. Continuous

H. Engine System Weight

I. Environment

[redacted]

SPECIFICATIONS OF RECOMMENDED PROPULSION SYSTEMS

In the analyses of Sections 4, 5, and 6, certain propulsion systems were recommended for use in the particular space missions considered. Using the previous specification catalog as a guide, the specifications of these propulsion systems are listed. The first and second stage propulsion systems for the lunar landing and return mission (using an intermediate orbit), and the Mars orbit establishment mission are described as well as the propulsion system recommended for the orbital rendezvous mission. These descriptions (Table 3-21 to 3-25) will provide useful propulsion system information in addition to illustrating the specification catalog. The specifications should be considered as preliminary. Further studies may indicate that some modifications are desirable.

[redacted]

TABLE 3-21

SPACE PROPULSION SYSTEM SPECIFICATIONS

Mars Orbit Establishment Vehicle

General Propulsion System Description

I. Energy Requirements

A. Total Impulse Required

1. Maximum = 3.3993×10^7 lb-sec

2. Minimum = 2.0952×10^7 lb-sec

B. Maximum Impulse

1. Increment = 3.2907×10^7 lb-sec

2. Mission: Mars Intermediate Orbit Establishment

C. Minimum Velocity Increment

1. Increment $\approx$ 0 lb-sec

2. Mission: Mars Intermediate Orbit Correction

D. Number of Increments = 3

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.2470 - 0.3093

b. Absolute Value = 30,000 lb

[redacted]

TABLE 3.21

(Continued)

2. Tolerance

   a. Engine-to-engine: ± 3.0 percent

   b. Run-to-run: ± 1.0 percent

3. Throttling

   a. Step: 3:1

   b. Continuous: None

4. Number of Restarts: 2

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

   1. Nominal: 5 (O/F)

   2. Tolerance: ± 0.5 percent

C. Specific Impulse

   1. Reference Engine Parameters

      a. Mixture Ratio: 5 (O/F)

      b. Chamber Pressure 500 psia

      c. Expansion Ratio 30

   2. Nominal Specific Impulse at Reference Conditions: 428 sec

[redacted]

TABLE 3-21

(Continued)

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start. Possible venting requirement.

2. Number of zero gravity engine starts: 3

4. Tank Venting: To relieve propellant heating problem.

B. Space Storage of Propellants

1. Environment: Earth-to-Mars vicinity

2. Storage Time: 250 days

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limit of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

[redacted]

[redacted]

TABLE 3-21

(Continued)

C. Component Design Restrictions: Protect from, or Design for Earth-to-Mars vicinity.

F. System Purging Requirements:

1. Number of Purges: 2

[redacted]

[redacted]

TABLE 3-21

SPACE PROPULSION SYSTEM SPECIFICATIONS

Orbital Establishment and Rendezvous Vehicle

General Propulsion System Description

I. Energy Requirements

A. Total Ideal Velocity Increment Required = 2700 fps

B. Maximum Velocity Increment

1. Increment = 2200 fps

2. Mission: 5 deg plane change

C. Minimum Velocity Increment

1. Increment = 1 fps

2. Mission: Rendezvous

D. Number of Increments

1. Maximum = 4

2. Minimum = 2

E. Maximum Cutoff Impulse Velocity Uncertainty = 0.5 fps

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.1

b. Absolute Value = 12,000 lb

[redacted]

TABLE 3-21

(Continued)

2. Tolerance

   a. Engine-to-Engine: ± 3 percent

   b. Run-to-Run: ± 1 percent

3. Throttling

   a. Step: None

   b. Continuous: None

4. Number of Restarts: 3

III. Propellants

A. Composition: Mixed Oxides of Nitrogen/Monomethylhydrazine

B. Mixture Ratio

   1. Nominal: 2.4 (O/F)

   2. Tolerance: ± 0.5 percent

C. Specific Impulse

   1. Reference Engine Parameters

      a. Mixture Ratio: 2.4 (O/F)

      b. Chamber Pressure: 150 psia

      c. Expansion Ratio: 25

   2. Nominal Engine Specific Impulse at Reference Conditions: 317 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

   1. Liquid/Vapor Separation Requirement: Provide liquid for engine start.

[redacted]

TABLE 3-21

(Continued)

2. Number of Zero Gravity Engine Starts: 4

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth Vicinity

2. Storage Time: 1 hr. to 1 day (depending on landing requirements)

3. Propellant Temperature Limits

a. Mixed Oxides of Nitrogen

(1) Lower: Freezing (-76 F)

(2) Upper: Propellant vapor pressure and density shall not exceed limits of propellant tanks and engine.

b. Monomethylhydrazine

(1) Lower: Freezing (-63 F)

(2) Upper: Vapor pressure and density shall not exceed propellant tank or engine limits.

C. Component Design Restrictions: Design for operation in earth vicinity space environment or provide protection from the environment.

F. System Purging Requirements:

1. Number of Purges: 3

System Component Requirements

I. Airframe and Propellant Tanks

A. Propellants

[redacted]

[redacted]

TABLE 5-21

(Continued)

1. Propellant Description

   a. Propellants: Mixed Oxides of Nitrogen/Monomethylhydrazine

   b. Nominal Mixture Ratio: 2.4 (O/F)

2. Useable Propellant Weight: 28,000 lb

3. Reserve Propellant Weight

   a. Flight Performance: 280 lb

   b. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 8 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purposes of Pressurization: Provide energy for expelling propellants from tank into combination chamber

B. Gas Volume in Propellant Tank

1. Increments: 4

[redacted]

[redacted]

TABLE 5-21

(Continued)

F. Environment

1. Storage

a. Time: 1 Day Maximum

2. Thermal: Earth Vicinity

III. Engine System

A. Propellant Description

1. Propellants: Mixed Oxides of Nitrogen/Monomethylhydrazine

3. Mixture Ratio

a. Nominal: 2.4 (O/F)

b. Tolerance: ± 0.5 percent

B. Thrust

1. Nominal: 12,000 lb

2. Tolerance

a. Run-to-Run: ± 1 percent

b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Pressurized Gas

D. Specific Impulse

1. Nominal: 317 sec

G. Throttling Requirement

1. Step: None

2. Continuous: None

I. Environment: Earth Vicinity

[redacted]

[redacted]

TABLE 3-23

SPACE PROPULSION SYSTEM SPECIFICATIONS

Lunar Landing and Return Vehicle (Orbital): Stage 1

ENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

A. Total Ideal Velocity Increment Required = 13,700 fps

B. Maximum Velocity Increment

1. Increment = 10,150 fps

2. Mission: Earth/Moon Transfer

C. Minimum Velocity Increment

1. Increment = 150 fps

2. Mission: Mid-course Correction

D. Number of Increments = 4

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.35

b. Absolute Value = 125,000 lb

2. Tolerance

a. Engine-to Engine: ± 3%

b. Run-to-Run: ± 1%

3. Throttling

a. Step: None

b. Continuous: None

[redacted]

[redacted]

TABLE 3-23

(Continued)

4. Number of Restarts: 3

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

1. Nominal: 5.0 (O/F)

2. Tolerance: ± 0.5 percent

C. Specific Impulse

1. Reference Engine Parameters

a. Mixture Ratio: 5.0 (O/F)

b. Chamber Pressure: 500 psia

c. Expansion Ratio: 30

2. Nominal Engine Specific Impulse at Reference Conditions: 428 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start.

2. Number of Zero Gravity Engine Starts: 4

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth/Moon Vicinity

2. Storage Time: 4-5 days

[redacted]

[redacted]

TABLE 3-25

(Continued)

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

E. Component Design Restrictions: Protect from, or Design for, Earth/Moon Vicinity Space Environment.

F. System Purging Requirements:

1. Number of Purges: 3

[redacted]

TABLE 5-25

(Continued)

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5.0 (O/F)

2. Useable Propellant Weight: 223,150 lb

3. Reserve Propellant Weight

a. Flight Performance: 2,230 lb

c. Trapped: 3, 060 lb

d. Fuel Bias: 1,225 lb

e. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 3 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for 4 engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purposes of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

[redacted]

TABLE 3-23

(Continued)

B. Gas Volume in Propellant Tank

1. Increments: 4

E. Environment

1. Storage

a. Time: 4 to 5 days

2. Thermal: Earth/Moon Vicinity

III. Engine System

A. Propellant Description

1. Propellants: Liquid Oxygen/Liquid Hydrogen

3. Mixture Ratio

a. Nominal: 5.0 (O/F)

b. Tolerance: ± 0.5 percent

B. Thrust

1. Nominal: 125,000 lb

2. Tolerance

a. Run-to-Run: ± 1 percent

b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Turbopump

D. Specific Impulse

1. Nominal: 428 sec

G. Throttling Requirement

1. Step: None

2. Continuous: None

I. Environment: Earth/Moon Vicinity Space

[redacted]

TABLE 3-24

SPACE PROPULSION SYSTEM SPECIFICATIONS

Lunar Landing and Return Vehicle (Orbital): Stage 2

GENERAL PROPULSION SYSTEM DESCRIPTION

I. Energy Requirements

A. Total Ideal Velocity Increment Required = 15,770 fps

B. Maximum Velocity Increment

1. Increment = 6400 fps

2. Mission: Landing from Orbit

C. Minimum Velocity Increment

1. Increment = 60 fps

2. Mission: Elliptical Orbit Establishment

D. Number of Increments = 6

II. Thrust

A. Magnitude

1. Steady-state Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.68

b. Absolute Value = 77,500 lb

2. Tolerance

a. Engine-to-Engine: ± 3 percent

b. Run-to-Run: ± 1 percent

3. Throttling

a. Step: 6:1

b. Continuous: 6 percent

TABLE 5-25

(Continued)

4. Number of Restarts: 5

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

1. Nominal: 5.0 (O/F)

2. Tolerance: ± 0.5 percent

C. Specific Impulse

1. Reference Engine Parameters

a. Mixture Ratio: 5.0 (O/F)

b. Chamber Pressure: 500 psia

c. Expansion Ratio: 30

2. Nominal Specific Impulse at Reference Conditions: 428 sec

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start

2. Number of zero gravity engine starts: 5

4. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth/Moon Vicinity

2. Storage Time: 2 Weeks

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

[redacted]

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

C. Component Design Restrictions: Protect from, or Design for, Earth/Moon Vicinity Space Environment.

F. System Purging Requirements:

1. Number of Purges: 5

[redacted]

[redacted]

TABLE 3-24

(Continued)

SYSTEM COMPONENT REQUIREMENTS

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5.0 (O/F)

2. Useable Propellant Weight: 76,150 lb

3. Reserve Propellant Weight

a. Flight Performance: 762 lb

c. Trapped: 1,040 lb

d. Fuel Bias: 418 lb

e. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

3. Atmosphere Flight: 3 g Axial

4. Space Flight: 4 g Axial

E. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellants for 5 engine starts.

2. Tank Venting: None

II. Pressurization System

A. Purpose of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

[redacted]

TABLE 3-24

(Continued)

B. Gas Volume in Propellant Tank

1. Increments: 6

E. Environment

1. Storage

a. Time: 2 Weeks

2. Thermal: Earth/Moon Vicinity

III. Engine System

A. Propellant Description

1. Propellants: Liquid Oxygen/Liquid Hydrogen

3. Mixture Ratio

a. Nominal: 5.0 (O/F)

b. Tolerance: ± 0.5 percent

B. Thrust

1. Nominal: 77,500 lb

2. Tolerance

a. Run-to-Run: ± 1 percent

b. Engine-to-Engine: ± 3 percent

C. Type of Feed System: Turbopump

D. Specific Impulse

1. Nominal: 428 sec

G. Throttling Requirement

1. Step: 6:1

2. Continuous: 6 percent

I. Environment: Earth/Moon Vicinity Space

[redacted]

TABLE 3-25

SPACE PROPULSION SYSTEM SPECIFICATIONS

Mars Orbit Establishment Vehicle, First Stage

General Propulsion System Description

I. Energy Requirements

A. Total Impulse Required

1. Maximum = 9.8766×10^7 lb-sec.

2. Minimum = 8.850×10^7 lb-sec.

B. Maximum Impulse:

1. Increment = 9.8766×10^7 lb-sec.

2. Mission: Earth Orbit Departure

C. Minimum Impulse

1. Increment = 8.850×10^7 lb-sec.

2. Mission: Earth Orbit Departure

D. Number of Increments = 1

II. Thrust

A. Magnitude

1. Steady-State Design Thrust Magnitude

a. Initial Thrust-to-Earth Weight Ratio = 0.4237

b. Absolute Value = 150,000 lb.

[redacted]

[redacted]

TABLE 3-25

(Continued)

2. Tolerance

   a. Engine-to-engine: ±3.0 percent

   b. Run-to-run: ±1.0 percent

3. Throttling

   a. Step: None

   b. Continuous: None

4. Number of Restarts: 0

III. Propellants

A. Composition: Liquid Oxygen/Liquid Hydrogen

B. Mixture Ratio

   1. Nominal: 5 (o/f)

   2. Tolerance: ±5 percent

C. Specific Impulse

   1. Reference Engine Parameters

      a. Mixture Ratio: 5 (o/f)

      b. Chamber Pressure: 500 psia

      c. Expansion Ratio: 30

   2. Nominal Specific Impulse at Reference Conditions: 428 sec.

[redacted]

TABLE 3-25
(Continued)

IV. Environmental Restrictions

A. Zero Gravity Propellant Supply

1. Liquid/Vapor Separation Requirement: Provide liquid propellant for engine start.

2. Number of Zero Gravity Engine Starts: 1

3. Tank Venting: None

B. Space Storage of Propellants

1. Environment: Earth Vicinity

2. Storage Time: A few days

3. Propellant Temperature Limits

a. Liquid Oxygen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limit of propellant tank and engine.

b. Liquid Hydrogen

(1) Lower: Propellant Freezing

(2) Upper: Propellant vapor pressure and density must not exceed limits of propellant tank and engine.

C. Component Design Restrictions: Protect from or Design for Earth Vicinity Environment.

D. System Purging Requirements:

1. Number of Purges: None

TABLE 3-23

(Continued)

System Component Requirements

I. Airframe and Propellant Tanks

A. Propellants

1. Propellant Description

a. Propellants: Liquid Oxygen/Liquid Hydrogen

b. Nominal Mixture Ratio: 5 (o/f)

2. Useable Propellant Weight:

a. Maximum: 235,500 lb.

b. Minimum: 211,780 lb.

3. Reserve Propellant Weight

a. Flight Performance: 2,355 lb. (maximum)

b. Trapped: 3,220 lb. (maximum)

c. Fuel Bias: 1,290 lb. (maximum)

d. Boil-off: None

B. Tank Loads

1. Handling: 4 g Lateral

2. Atmosphere Flight: 3 g Axial

3. Space Flight: 4 g Axial

C. Zero Gravity Requirements

1. Gas/Liquid Separation: Provide liquid propellant for engine st

2. Tank Venting: None

[redacted]

[redacted]

TABLE 3-25
(Continued)

II. Pressurization System

A. Purposes of Pressurization: Provide sufficient NPSH for turbopump operation and assist in providing structural support as required.

B. Gas Volume in Propellant Tank

1. Increments: 1

C. Environment

1. Storage

a. Time: a few days

2. Thermal: Earth vicinity

III. Engine System

A. Propellant Description

1. Propellants: Liquid Oxygen/Liquid Hydrogen

2. Mixture Ratio

a. Nominal: 5 (o/f)

b. Tolerance: ±0.5 percent

B. Thrust

1. Nominal: 150,000 lb.

2. Tolerance

a. Run-to-Run: ±1.0 percent

b. Engine-to-Engine: ±3.0 percent

[redacted]

TABLE 3-25

(Continued)

C. Type of Feed System: Turbopump

D. Specific Impulse

1. Nominal: 428 sec.

E. Throttling Requirement

1. Step: None

2. Continuous: None

F. Environment: Earth Vicinity

[redacted]

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